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NASA - INDUSTRY APOLLO TECHNICAL CONFERENCE

WASHINGTON, D.C.
July 18, 19, 20, 1961

A COMPILATION OF THE PAPERS PRESENTED

PART II
July 20

[redacted]

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
WASHINGTON

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DECLASSIFIED

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INTRODUCTION

This document is Part II of a compilation of papers presented at an NASA-Industry Apollo Technical Conference held at the Interdepartmental Auditorium, Washington, D.C., July 18, 19, and 20, 1961. These papers were presented by staff members from NASA Centers and personnel of the NASA Apollo Study Contractors. These contractors were General Dynamics/Astronautics, General Electric (Missile Systems Vehicle Division), and The Martin Company.

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VIII. PROJECT MERCURY MISSION AND DESIGN

MERCURY ORBITAL MISSION

By Charles W. Mathews and Gerald W. Brewer

NASA Space Task Group

The objectives of the Mercury orbital mission are well known, but many of the technical and operational implications of the basic ground rules may not be as well understood. Of particular importance is the fact that man is an intrinsic part of the Mercury spacecraft system. Therefore, all of the associated flight-safety ramifications are inherent in the program. Somewhat paradoxically, timeliness has necessitated a high degree of integration of hardware and technology developed in connection with unmanned missile programs. A subsequent paper by F. J. Bailey, Jr., and John C. French describes some details of the approach to flight safety and reliability in the Mercury program in the light of these and other constraints.

The purpose of the present paper is to discuss broadly the evolvement of the Mercury mission. This discussion is intended to provide background for understanding of the more detailed papers on each phase of the Mercury program.

The objective of the Mercury orbital mission is to achieve manned orbital flight and recovery for the primary purpose of an initial determination of man's capabilities and functions in space.

Certain basic ground rules were set down at the outset of the program: New developments were to be minimized in order to rapidly achieve the objective. The simplest and most reliable approach was to be used, and, in order to obtain operational experience, flight-system qualification, and flight-crew training, a progressive buildup of tests was planned.

The specific approach taken under these basic ground rules is indicated by the following items:

Atlas launch vehicle (propulsion and guidance)

Automatic escape system

Unmanned and animal flights

Ground monitoring and in-flight control

Extensive pilot control capability

[redacted]

Retrorocket return with drag reentry

Parachute landing system

Water landing (primary)

The decision was made to use the Atlas as the launch vehicle for the orbital flights. It was the only vehicle well into its development flight program that had the performance and guidance accuracy required for this type of mission. At the same time the primary operational mission of the Atlas was such that it would not achieve a mission reliability compatible with a manned vehicle. Therefore, a highly reliable automatic escape system had to be evolved. The knowledge that certain catastrophic failures of the Atlas could develop too rapidly for timely human action resulted in incorporation of an automatic abort sensing and implementation system as a part of the escape-system concept. Details of this system and some other aspects of the launch-vehicle—spacecraft compatibility are presented in a subsequent paper by Tecwyn Roberts, Paul C. Donnelly, and Arthur Jonas.

The Mercury orbital mission is a very big step in manned flight operations and is not completely amenable to the usual type of flight-test buildup in terms of speed, altitude, and flight duration. For this reason a flight-test program was evolved which included unmanned and animal flights preceding manned flights for each of the major missions. This approach does produce a severe technical task in that the Mercury system had to be designed for completely automatic operation while incorporating all of the support and backup functions associated with the manned operation. Some of the engineering factors relating to these crew considerations are discussed in a paper by Richard S. Johnston and Gerard J. Pesman.

Worldwide ground monitoring using real time communications is provided for flight-safety purposes during manned flights and for mission control during unmanned flights. Even in certain phases of the manned flights critical functions must be exercised from the ground.

Although the Mercury spacecraft is capable of completely automatic flight under ground control, the aim was to provide the astronaut with extensive capability for system operation and management. This intent not only reflects the desire to achieve the objective of investigating man's capability but also to utilize the man in improving the probability of mission success.

The last three items on the list previously mentioned are associated with the need to achieve a simple, direct, and proven approach to the problem of return from orbit. The water-landing approach also is in recognition of the problems of landing after aborted launches and the extensive existing capability within the Navy for supporting a water search and recovery.

A description of the Mercury spacecraft is the subject of the next paper by Aleck C. Bond. Bond's paper and others on spacecraft details supply some insight as to how the general approach outlined herein was incorporated in the design.

The actual Mercury orbital flight is now described. The ground track for the mission is shown in figure 1. The vehicle is launched from Cape Canaveral on a heading slightly north of east; the insertion point occurs approximately halfway between Cape Canaveral and Bermuda. The design mission comprises three orbits with the retrofire operation taking place off the west coast of the United States. The landing occurs in the Atlantic Ocean north of Puerto Rico.

This particular track was chosen for the following reasons:

(a) The end of each of the first three orbits occurs over the continental United States.

(b) The track lies entirely within friendly territories and within the temperate region of the world.

(c) It makes good use of range facilities established prior to Project Mercury.

The primary station of the Pacific Missile Range is in a good position to monitor the critical retrofire operation, and the terminal phase of the landing can be monitored along the Atlantic Missile Range.

The major events that occur throughout the flight are indicated in figure 2. All three Atlas engines are fired simultaneously, but the vehicle is held down briefly to check for proper engine operation. Upon release, the Atlas rolls immediately in order to establish the desired launch azimuth. It proceeds vertically for a number of seconds and then a pitchover maneuver is programed into the autopilot. This slow maneuver continues, and at about $2\frac{1}{2}$ minutes after launch a staging operation occurs - that is, the two outboard engines of the Atlas are shut down and jettisoned in order to obtain increased performance. Shortly thereafter the spacecraft escape tower is also jettisoned for the same reason.

After staging the Atlas is precisely guided by means of a ground guidance and command system, and normally 5 minutes after lift-off orbit-insertion conditions are established about 427 nautical miles from the launch site at an altitude of 87 nautical miles. Although the flight-path angle is near zero at this time, the Atlas is cut off slightly above circular-orbit speed so that an apogee altitude of about 137 nautical miles is achieved.

After thrust cut-off, the spacecraft is automatically released from the launch vehicle, and small posigrade rockets are fired to produce separation. The spacecraft then maneuvers to retroattitude (blunt face forward) in order to be ready to retrofire in case immediate return is necessary. It is held in this attitude by the autopilot during orbital flight except when the astronaut chooses to maneuver to other attitudes. It should be noted at this point that the astronaut has complete backup capability in initiating the onboard sequence of operations should the automatic systems fail.

If an abort of the mission is necessary during powered flight, it can be initiated by the pilot, by a ground-command radio-frequency (RF) link, or in certain cases of catastrophic launch-vehicle failure by an automatic system as previously mentioned. If the abort signal occurs prior to staging as shown in figure 3, the spacecraft is released and a large solid rocket motor atop the escape tower produces rapid and stable separation of the spacecraft from the launch vehicle. The escape tower is then jettisoned at maximum altitude; later a landing parachute is deployed much in the same manner as that which occurs at the termination of a normal orbital flight. If the abort signal occurs after staging, the vehicle is for all practical purposes out of the atmosphere and the spacecraft—launch-vehicle separation is accomplished with the same posigrade-rocket sequence as for a normal insertion.

Retrofiring for return from orbit can be accomplished directly by the pilot, by RF ground command, or by an onboard timing device which can be set by the pilot or by ground personnel. For a normal three-orbit misssion the retrofire operation takes place about 275 minutes after launch. In any case the time of retrofire is precisely and rapidly determined by computers from data received from a worldwide net of tracking stations. After retrofire the retropackage is automatically jettisoned and the spacecraft is maneuvered into reentry attitude. Reentry into the atmosphere occurs about 10 minutes after retrofire and upon decelerating to subsonic speeds a drogue parachute is deployed which aids the autopilot in stabilizing the spacecraft. It also serves to pull out the main parachute at an altitude of 10,000 feet. Certain recovery aid devices are activated at this time, and shortly thereafter the shock-absorbing landing bag is deployed.

On landing, the main parachute is jettisoned, and the pilot establishes communications with the recovery forces. The landing takes place about 23 minutes after retrofire, and the total distance covered during reentry is about 3,150 nautical miles.

Some of the actual operational activities involved in carrying out the Mercury mission are as follows:

Vehicle checkout

Network checkout

Preflight-crew activities (astronauts and flight controllers)

Weather forecasting

Scheduling

Flight-safety activities

Medical operations

Launch operations

Network operations

Flight-control operations

Recovery operations

Postmission analysis and reporting

All of these activities involve mission planning, facilities implementation, and personnel training. Such planning activities involve all aspects of the operation and commenced quite early in the project; however, most of the direct support to the flight takes place during the five main mission phases shown in the following table:

Prelaunch	Launch	Flight	Recovery	Postrecovery
Spacecraft tests	Countdown	Status reporting	Landing prediction	Return to base
Launch-vehicle tests	Servicing	Flight monitoring	Active search	Debriefing
Pad checks	Checking	Control action (a) Normal (b) Emergency	Pickup	Data analysis
Forces deploy	Status reporting			Reporting

The prelaunch phase involves spacecraft and launch-vehicle tests in hangar facilities, readiness checks at the launch complex, and deployment of support elements, such as flight monitoring teams and recovery forces, to remote operational locations. The launch phase proceeds with a countdown wherein servicing and checking are accomplished in a synchronized manner designed to bring all elements to launch readiness at the same time. These activities are discussed in a subsequent paper by G. Merritt Preston and Dugald O. Black.

The actual flight involves many activities. Status information is exchanged between astronaut and ground stations and, in turn, this information is used to update a central point of mission control. The flight systems are monitored by the astronaut and flight controllers on the ground, and normal-mission control functions or emergency actions are initiated where required. Details of these activities are discussed in a subsequent paper by Christopher C. Kraft, Jr., and C. Frederick Matthews.

Recovery involves dispensing of landing-location information to the forces involved, active electronic search by aircraft, and pickup of astronaut and spacecraft by helicopter or ship.

The postrecovery phase involves return of the astronaut and spacecraft to a preselected staging point and the necessary debriefing, analysis, and reporting so that the experience of the flight can be integrated into planning for subsequent missions.

As already implied, ground monitoring and control of the flight are accomplished by means of a network of stations located throughout the world (17 in all). The distribution of these stations, as shown in figure 4, was established on the basis of certain tracking, communication, command, and telemetry requirements. Continuous voice and telemetry contact with the spacecraft was desired during the powered flight and for a 15-minute period thereafter. The same requirement of continuous contact applies to the reentry phase - not only for the end of orbital flight but also for any reentry resulting from aborts during the powered phase of flight. The distribution of stations over the rest of the ground track is such that no unplanned reentry could occur without the knowledge of at least one station. There are a few exceptions to this capability along the third orbit, and on the basis that whenever the mission has progressed satisfactorily into the third orbit, some relaxation of contact requirements is justified.

The major systems located at each station are given in table I. Air-to-ground voice and telemetry are available at most sites except in cases where overlapping coverage does not warrant the installation. Radar systems are distributed to cover continuously the launch, launch abort, and all reentry cases and are also sited in Hawaii and Australia for use in precise orbit determination. Command stations are located wherever a ground command function is needed for initiation of mission aborts, for initiation of normal reentry, or for timer reset. As exists for the onboard systems, complete backup capability is incorporated in the network systems.

All remote stations are connected by ground communications to two major facilities: a control center at Cape Canaveral and a computing and communications center at the Goddard Space Flight Center near Washington, D.C. (See fig. 5.) The control center is the focal point of activities as regards to mission status. Generally, ground command decisions are made at this center although such decisions may be relayed to other stations for action. During the flight the control center generally is in communication with the station having the spacecraft overhead in order to be rapidly advised on status.

In addition to being a very large communications switching point supporting the control-center activities, the communications and computing center processes all radar data required to establish trajectory information, including display data, acquisition data, landing prediction data, and the like. Two modes of operation are used. During powered flight, data from the Cape Canaveral radars are sent back and forth along high-speed lines to be processed at Goddard and displayed at Cape Canaveral. Data from radars at remote sites are also transmitted and processed automatically; however, teletype is the communications media and therefore the processing is at relatively lower speeds.

Now the basic concept of the recovery support to Project Mercury is discussed. A positive recovery plan exists for all possible conditions of landing whether the landing results from firing the escape rocket while on the launch pad or from initiation of a reentry at some arbitrary point along the ground track. This plan is characterized by electronic search by aircraft for direction-finding (DF) signals from active beacons aboard the spacecraft. Ship recovery forces are deployed in planned recovery areas and when necessary for recovery are vectored by the locating aircraft. The amount and types of aircraft and ship support provided in any given area are a function of the probability of occurrence of a landing in that area.

The types of recovery situations that have been considered are illustrated in figure 6. There are local-area recovery situations resulting from aborts off the pad, or shortly after lift-off. Another recovery situation results from a water landing caused by aborts prior to staging. This situation has a reasonable probability of occurrence but the extent of the recovery area is relatively small. Aborts after staging are of lower probability but potentially cover all of the broad Atlantic Ocean area along the launch ground track. Special retrofiring procedures are necessary in this instance in order to limit the extent of the recovery support and to constrain the landing area to that shown.

Once a satisfactory orbit has been attained there is a high probability that a normal end of orbit reentry will take place because of the high system redundancy onboard the spacecraft; however, double-failure situations, such as stuck thrusters in both the normal or automatic attitude control systems, could force an early reentry. For this reason, preferred contingency areas have been established as shown in this figure, and plans have been made for controlling reentries into these areas and for recovery in the event such a situation does occur. Details of the ground rules, procedures, and type and nature of the support are given in a subsequent paper by Robert F. Thompson, William C. Hayes, Jr., and Donald C. Cheatham.

In interpreting this discussion of the general aspects of the flight plan for the Mercury orbital mission, it should be remembered that this mission is an initial venture into manned space flight. Undoubtedly some of the concepts will change as more flight experience is obtained. Nevertheless, it is presently possible to summarize in a useful manner many aspects of the Mercury approach. The detailed papers which follow will attempt to accomplish this purpose in relation to the projected Apollo mission.

TABLE I

MERCURY GROUND STATIONS

		COMM. & T/M	S OR C RADAR	COM'D. CAPAB.
CAPE CANAVERAL	(CNV)	X	(S)-C	X
BERMUDA	(BDA)	X	S-C	X
ATLANTIC OCEAN SHIP	(ATS)	X	-	-
NORTHWEST AFRICA	(CYI)	X	S	-
SOUTHWEST AFRICA	(KNO)	X	-	-
SOUTHEAST AFRICA	(ZZB)	X	-	-
INDIAN OCEAN SHIP	(IOS)	X	-	-
WEST AUSTRALIA	(MUC)	X	S	X
WOOMERA, AUSTRALIA	(WOM)	X	C	-
CANTON ISLAND	(CTN)	X	-	-
HAWAII	(HAW)	X	S-C	X
SOUTH CALIFORNIA	(CAL)	X	S-C	X
PACIFIC COAST	(GYM)	X	S	X
WHITE SANDS	(WHS)	-	C	-
SOUTH TEXAS	(TEX)	X	S	-
EGLIN (AFATC)	(EGL)	-	C	-
GRAND TURK	(GTI)	X	(S)	-

PROJECT MERCURY NOMINAL ORBIT

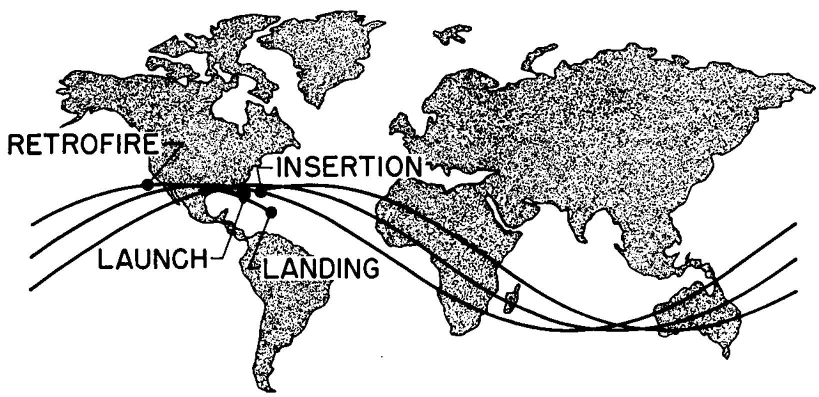

Figure 1

FLIGHT PLAN MERCURY-ATLAS

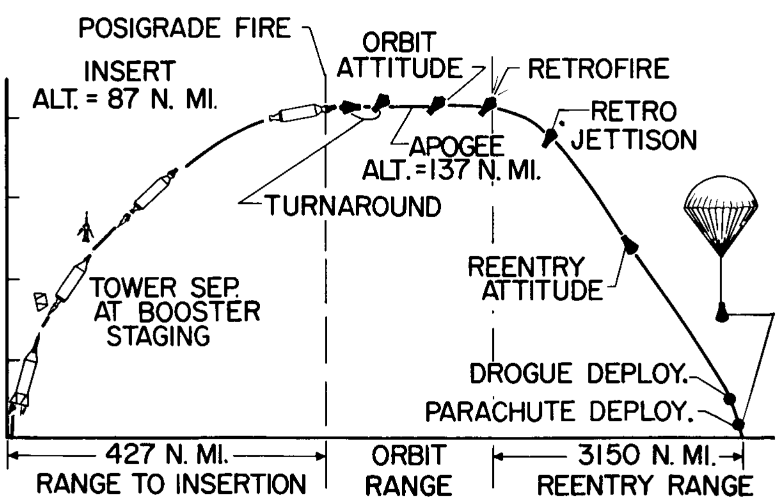

Figure 2

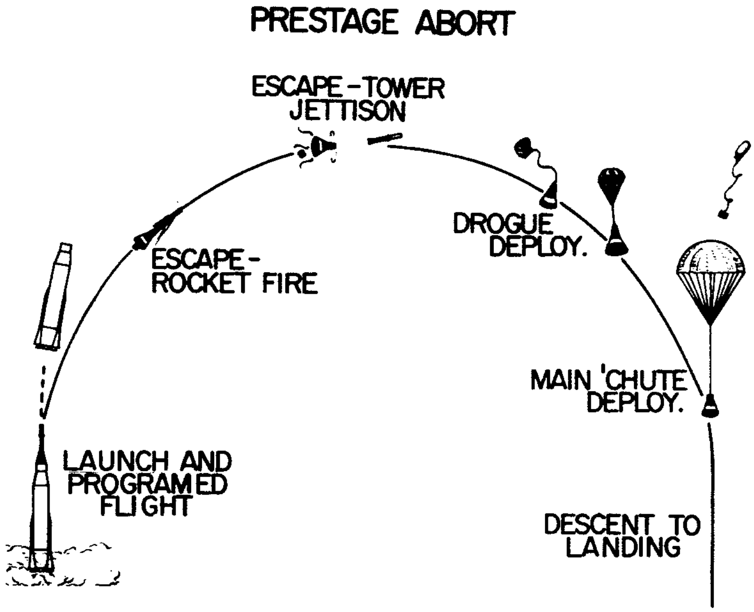

Figure 3

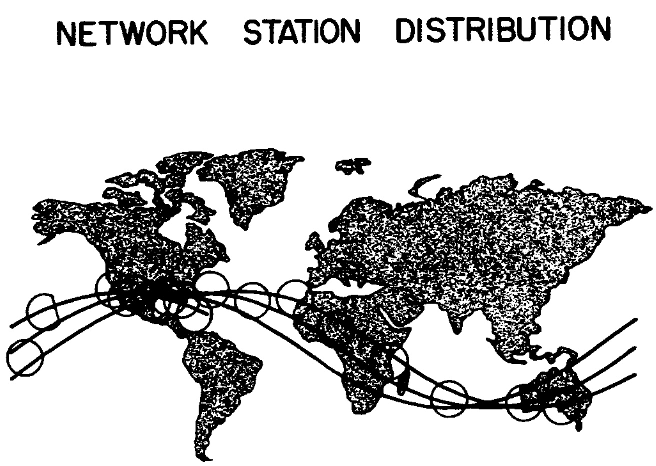

Figure 4

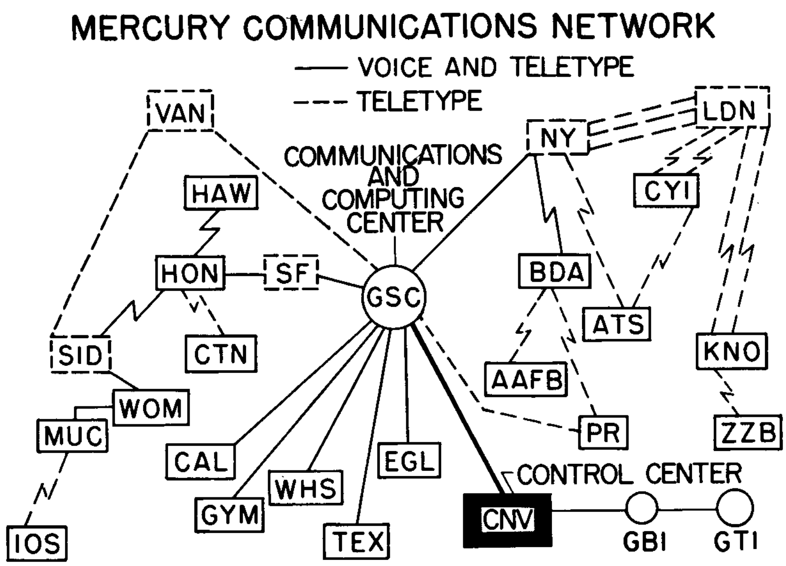

Figure 5

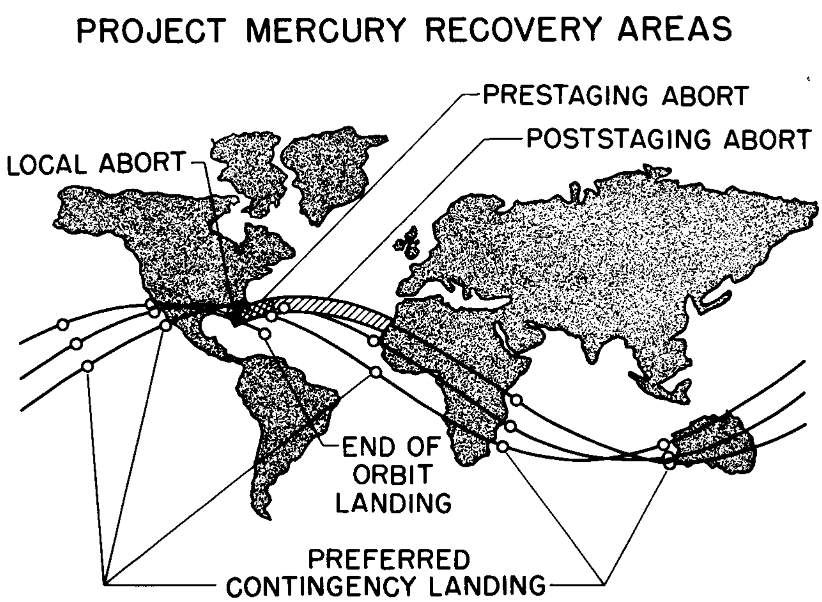

Figure 6

MERCURY SPACECRAFT DESCRIPTION

By Aleck C. Bond

NASA Space Task Group

INTRODUCTION

During the course of Project Mercury a number of unclassified papers have been published which have provided a general overall description of the Mercury spacecraft and its systems and the basic design concepts employed. (See refs. 1 to 6.) These papers have of necessity treated certain aspects of the spacecraft design in a rather superficial fashion; however, descriptions of most systems and their operation have been rather complete. It is therefore the purpose of this paper to present a brief review of the Mercury spacecraft design with some emphasis in areas which have not been covered in previous publications. In addition, some information has been included on certain aspects, such as heat protection and weight which should be of interest in the design of the Apollo vehicle.

SPACECRAFT CONFIGURATION

Figure 1 shows a sketch of the Mercury spacecraft with and without its escape system. The overall length of the vehicle, including the escape tower and retropackage, is 311.55 inches. The maximum diameter of the spacecraft is 74.5 inches. The basic-spacecraft configuration is characterized by certain features: the blunt reentry face, which has a radius of curvature of 80 inches; the conical afterbody, which has a half-angle of 20°; the cylindrical recovery compartment, which has an external diameter of 32 inches; and the trapezoidal antenna canister, which is 24.50 inches in length. An overall view of the spacecraft can be seen in figure 2. This figure shows the spacecraft mounted in the weight and balance fixture during the determination of the lateral location of the center of gravity.

Considerations for low as well as uniform heating to the heat shield determined the radius of curvature of the blunt face. The afterbody shape was dictated by considerations of both heating and stability for the reentry portion of flight. The inward-sloping surfaces of the cone tend to minimize the afterbody heating, and the extensions to the cone enhance both the static and dynamic stability, as well as provide storage space for the landing and recovery systems and certain control-system equipment.

The orbit configuration, which is shown in the right-hand side of figure 1, has an overall length of 131.57 inches. The length of the reentry configuration, which is the orbit configuration less the retropackage, is 115.06 inches.

ESCAPE AND RETROGRADE SYSTEMS

The escape tower is attached to the recovery-compartment structure by means of a three-segment Marman-type clamping band which is held together by means of explosive bolts. The solid-propellant escape rocket mounted on top of the tower is designed to provide an adequate separation distance between spacecraft and launch vehicle in case of launch-vehicle failure. The nominal sea-level impulse of this motor is 56,500 pound-seconds and it has a burning time of about 1.4 seconds. If the launch vehicle fails on the launch pad the escape rocket will lift the spacecraft to an altitude of approximately 2,500 feet, which is sufficient to allow deployment of the main parachute. The escape motor incorporates a triple nozzle; however, the resultant thrust line is canted slightly in order to provide some lateral displacement between the escaping spacecraft and the flight path of the launch vehicle. In an aborted mission a small solid-propellant rocket motor which is nestled among the triple nozzles of the escape motor is used to jettison the tower from the spacecraft. This motor has a nominal thrust of 785 pounds. In a normal mission the tower is jettisoned by firing the escape motor. Tests of the escape system, simulating an off-the-pad abort, an abort at maximum dynamic pressure, and an abort at very high altitude have demonstrated the soundness of the Mercury escape concept.

The retropackage, which is shown mounted to the heat shield in figure 1, contains six solid-propellant rocket motors, three being retrograde motors and the other three being posigrade motors. The metal retropackage provides micrometeorite and thermal shielding for the rocket motors. The retrograde motors which are used to initiate the reentry from orbit will provide a velocity change of 450 feet per second along the longitudinal axis of the spacecraft. The nominal thrust of these motors is 992 pounds and the burning time is 13.2 seconds. These motors are ripple fired at 5-second intervals. The posigrade motors, on the other hand, are all fired simultaneously and are used to effect separation of the spacecraft from the launch vehicle. These motors, which are considerably smaller than the retrograde motors, will impart to the spacecraft a velocity increment of 30 feet per second. These motors provide a nominal thrust of 420 pounds for 1 second in vacuum.

SPACECRAFT STRUCTURE AND HEAT PROTECTION

Entrance to the cabin, which is within the conical section, is gained through a hatch in the wall of the conical afterbody, shown in figures 1 and 3. Figure 3 also shows a closeup view of the observation window which is located just above the astronaut's head. The window allows the astronaut to make visual observations of his surroundings, independent of the existing optical system. Through this window he can also observe the functioning of the escape system, as well as the deployment of the parachutes. This window utilizes heat-resistant Vycor glass and is of multipane construction.

The spacecraft afterbody is of double wall construction with the walls separated with blankets of Thermoflex insulation. The inner wall is reinforced with longitudinal hat section stringers and forms the load-carrying structure of the spacecraft. In the region of the pressure compartment the stringers are seam-welded to the skins of the inner wall. Figure 4 shows four photographs which describe the internal spacecraft structure. Titanium is used for both the inner skins and the stringers. The pressure compartment inner wall, including the large bulkhead, is constructed of two layers of 0.010-inch titanium which are seam-welded together. The outer layer of skin is beaded in order to increase panel stiffness. The insulation blankets which are situated between the stringers can be seen installed in the photograph in the lower left of the figure.

Figure 5 shows a sketch which further illustrates the heat protection and wall construction utilized on the Mercury spacecraft. The conical afterbody and antenna canister are covered with overlapping shingles which are unrestrained for thermal expansion. These shingles are made of René 41 refractory alloy and are corrugated for stiffness. The thickness of the shingles is 0.016 inch on the conical afterbody and 0.031 inch on the antenna canister. These shingles are preoxidized to provide a high emissivity for thermal dissipation through radiation. The recovery compartment outer wall is constructed of a series of beryllium plate elements, 0.22 inch thick, which are also unrestrained for thermal expansion. Beryllium is used in this area in order to accommodate the high localized heat loads associated with flow reattachment, particularly at the high angles of attack. The blunt end of the spacecraft is protected from reentry heating by a heat shield. For the orbital missions an ablative-type heat shield is provided, whereas for the Redstone suborbital missions a heat shield constructed of beryllium is employed.

The antenna canister employs an 8-inch-long Vycor window which acts as a dielectric between the top of the canister and the remainder of the spacecraft. As previously mentioned, Vycor glass is also used to cover the large cabin-observation window which can be seen in the figure.

Figure 5 also shows typical sections of the conical afterbody and recovery compartment walls. These sections indicate the method of support of the outer shingles and the insulation arrangement. At the support points the shingles are thermally isolated from the inner structure by strips of Min-K insulation wrapped in Inconel foil. The Min-K strips are supported in thin fiber-glass channels which are attached to the stringers. The longitudinal stringers are also covered with a reflective foil tape in order to reduce the radiant heat transfer to the inner structure.

The construction of the ablation heat shield is illustrated in the section shown in the lower right of figure 5. The heat shield is constructed of fiber-glass cloth impregnated with CT 37-9X resin which is a modified phenolic resin. The total thickness of the shield is 0.95 inch and is composed of two types of layup. The outer laminate, which is termed the shingle or ablation laminate, is 0.65 inch in thickness. In this laminate the plies of glass cloth are swept back with respect to the local flow direction. The angle between the cloth plies and the local tangent to the heat shield is approximately 20°. This type of construction of the outer laminate allows the gaseous ablation products to escape into the boundary layer without danger of shield delamination. The inner laminate which is 0.30 inch thick utilizes parallel layup. It is the structural element of the heat shield and is used to absorb the reentry airloads and landing impact loads.

SPACECRAFT SYSTEMS

In addition to the items discussed in the foregoing sections, the Mercury spacecraft incorporates a number of major systems which include (1) communications, (2) attitude control, (3) environmental control, (4) electrical power, (5) explosive devices, (6) general cabin equipment, and (7) landing and recovery systems. Since all the systems cannot be covered in detail in this presentation, only certain features of some systems of special interest will be discussed. The reader is referred to references 2 to 7 for further information on some of these systems. One thing which should be noted at this point is that although all spacecraft systems have been designed for completely automatic operation, provisions have also been made for operation and control of the systems by the astronaut. In addition, redundancy has been designed into all the systems.

When all these systems and their subsystems are integrated within the spacecraft, the internal arrangement is essentially that shown in the sketch of figure 6. With this arrangement, the astronaut has about the same amount of room as in a typical fighter cockpit. The astronaut is shown seated in his contoured couch with his back to the heat shield.

During launch the small end of the spacecraft is pointed forward, but for reentry the orientation is reversed and the heat shield is pointed forward. This reversal in attitude simplifies the astronaut's support system since the couch is properly alined for both the acceleration and deceleration phases of flight.

Starting at the small end of the spacecraft, the following items can be distinguished: the antenna canister, the pitch and roll horizon scanners, the drogue parachute, the main and reserve parachute, the pitch and yaw jets and associated plumbing, the periscope and instrument panel, the side arm controllers, the various electronic packages, the observation window, and the many other items of equipment needed to carry out the Mercury mission. The environmental control system is located primarily beneath the astronaut's couch.

Attitude-Control System

One of the more complex systems of the spacecraft is the attitude-control system. Attitude can be controlled by a completely automatic system or by the astronaut through the use of a manual-control system. The automatic and manual systems are completely independent. In fact, the reaction-control forces for each of the two systems are provided by completely separate hydrogen-peroxide-fuel and jet-thruster systems. The automatic system utilizes two sets of thrust chambers: one of high torque for large error correction, for instance, during retrorocket firing, and the other for small error correction while in orbit.

Electrical signals generated by the control gyros of the automatic system are used to control its various solenoid-operated fuel valves. However, with the manual system the astronaut uses the right-hand controller, shown in figure 7, to manipulate directly the manual fuel-control valves through a series of mechanical linkages. Further manual control of the spacecraft is provided for the astronaut through the use of the "fly-by-wire" mode. In this mode the astronaut can control the solenoid valves of the automatic control system by means of a series of electrical limit switches incorporated in the right-hand controller.

The right-hand controller (fig. 7) is a three-axis controller which allows the astronaut to make simple control inputs by short hand movements. Fore-and-aft movements provide control in the pitch plane; side-to-side movements give roll inputs; the twisting of the controller about its vertical axis gives yaw or directional control. This type of hand controller incorporates the standard aircraft stick motions for pitch and roll control. The twisting motion for yaw control replaces the function of the conventional airplane rudder pedals. The left-hand controller, incidentally, is used to provide the astronaut with a quick means for initiating an abort[redacted]f the left handgrip will initiate the abort sequence. A simple locking feature is incorporated in the controller to prevent an inadvertent abort.

The modes of operation of the automatic stabilization and control system (ASCS) are best discussed in conjunction with the Mercury normal mission profile shown in figure 8. In the left-hand side of the figure the mission begins with the launch of the vehicle and proceeds to staging or release of the first-stage engines. Approximately 20 seconds after staging, the escape system is jettisoned in order to reduce the insertion weight. With the jettisoning of the tower, the automatic stabilization and control system is activated. At this time, sequence A, the roll and pitch gyros are slaved to the horizon scanners. At insertion into orbit, sequence B, the sustainer and vernier engines are cut off, the main clamp ring is released, and the posigrade rockets are fired to separate the spacecraft from the launch vehicle. At this point the control system maintains rate damping for a period of 5 seconds in order to minimize disturbances arising from the firing of the posigrade rockets.

The turnaround maneuver is then carried out in the yaw plane, and the spacecraft is oriented to the 34° retrofire attitude, as shown in sequence C. In this orientation mode the control system maintains the high-torque thrusters active for correction of large disturbances. After about 30 seconds when the orientation disturbances have been diminished, the control system switches to the orbit mode, sequence D. In this mode the attitude angle is also 34°; however the low torque thrusters are now used for attitude control.

Upon completion of the orbital mode, the retrofire command is given and the control system switches back to the orientation mode, sequence E, and holds the retroattitude of 34° throughout the firing of the retromotors. Sixty seconds after retrofire, the retropackage is jettisoned and the spacecraft is oriented to the reentry of $-1\frac{1}{2}^{\circ}$ as shown in sequence F.

As the spacecraft reenters the atmosphere and perceptible g-forces begin to be sensed, sequence G, the control system discontinues the attitude programing. It then introduces a steady roll of 10° to 12° per second to reduce the landing-point dispersion and also maintains the rate damping to prevent large oscillation buildup. This mode of control continues through descent on the drogue parachute. At main parachute deployment, sequence H, the control system is turned off and its remaining fuel is jettisoned.

[redacted]

[redacted]

Instrument Panel

The instrument panel for the orbital Mercury spacecraft is shown in figure 9. It was chosen to be discussed at this point since it provides a sort of summation of many of the spacecraft systems. The controls and displays shown on the panel are grouped according to function. The group on the left has various astronaut controls such as those concerned with the automatic- and manual-attitude-control systems and the retrorocket system. The two large handles in the center of the group are for decompression and repressurization of the cabin. Decompression would be the method used for extinguishing a fire. The small control panel shown in the upper left of the figure incorporates the cabin- and suit-temperature controls.

The next group is a sequential display consisting of a series of light indicators designed to tell the astronaut whether various functions occurred at the proper time. A green light will show that the function occurred and a red light will indicate a failure in the automatic system. The handle or switch just to the left of each indicator allows the astronaut to override and correct the failure of a given function. The circular light at the top of this group is the abort light which comes on in the event an abort is initiated. The switch located immediately below the abort light is a ready switch and is used during countdown to inform the test conductor of the astronaut's readiness for launch.

The next series of four circular dials read longitudinal acceleration, percent of fuel supply in the hydrogen peroxide tanks of both the automatic and manual control systems, rate of descent, and altitude. The combination display at the top center of the panel presents angular rate and spacecraft attitude data in each of the three axes. The rate display is in the center surrounded by the three attitude dials. The astronaut's control of spacecraft attitude will be aided by observations through the periscope. The astronaut also uses the periscope during descent to observe parachute deployment. The periscope screen can be seen in the lower center of the panel.

The instrument located above the left of the periscope screen is an earth-path indicator and it consists of a small-scale model of the earth which is synchronized to rotate in time with the orbit. To the right of this is the satellite clock, which in addition to programing the firing of the retrorockets, indicates the time of day, the lapsed time from launch, and time to go for retrofire.

The environmental control system display is grouped in the upper right-hand portion of the panel. This group indicates functional information on the system such as cabin pressure and temperature, suit pressure and temperature, relative humidity, and oxygen and coolant quantity. The electrical-power-system monitor dials and the communications controls are located directly below the environmental group. Just to the right of these two groups are a series of warning lights which will give the astronaut a visual warning in the event of the failure of a major system. This warning is further emphasized by an auditory warning signal. The astronaut may turn off the auditory signal by actuating the switch adjacent to the activated warning light and then take corrective action. A number of fuses of certain critical systems are located at the extreme right of the instrument panel.

ACCELERATION AND IMPACT ATTENUATION

Considerable attention has been given to the design and development of suitable acceleration and impact attenuation systems for the Mercury spacecraft. The accelerations to which the astronaut is subjected under normal exit and reentry conditions are an order of magnitude higher than those associated with conventional high-speed aircraft, but they are by no means the highest which may be encountered in the Mercury flight spectrum. The emergency-abort conditions actually present the more severe loading conditions. During certain escape maneuvers the astronaut can be subjected to accelerations as high as 17g as a result of firing the escape rocket. Reentry loads of the order of 20g can result if a mission is aborted some few seconds prior to insertion into orbit. The astronaut is protected from undue localized loadings by means of his form-fitting or contoured couch. Reverse loadings are absorbed by a restraint harness. Details of both the contoured couch and harness system are given in reference 7.

During the various spacecraft drop tests, it was found that impact on water under certain surface conditions could produce accelerations as high as 40g for a few milliseconds with average onset rates of about 8,000 to 10,000 g/second. Impact on land would produce even higher loadings. Some impact attenuation is provided by the aluminum honeycomb incorporated in the couch support structure; however, this was not adequate for all contingencies. In order to attenuate the impact accelerations further, particularly for conditions with attendant high surface winds, a simple air cushion or landing impact bag was devised as is shown schematically in figure 10. A photograph of the landing bag installation is shown in figure 11. The system consists of a 4-foot skirt made of rubberized fiber glass that is attached on the one end to the heat shield and on the other end to the spacecraft. After the main parachute is deployed, the heat shield is detached from the spacecraft structure, thereby allowing the skirt to extend and fill with air. Upon impact, the air trapped within the skirt is vented through the series of holes located in its upper and lower ends. A series of thin metal straps which are slightly shorter than the skirt are used to absorb the lateral impact loads and thus prevent damage to the skirt. Maintaining the integrity of the assembly is necessary since the extended skirt and shield combination provide an upright attitude and flotation stability when the spacecraft is in the water.

A recent series of drop tests with this system, with surface winds as high as 20 knots, have yielded measured impact accelerations no higher than 16.5g. The associated average onset rates were of the order of 200 g/second.

SPACECRAFT WEIGHTS

It is well known that weight is a most important factor in the design of a spacecraft. Weight control is also a difficult and delicate problem. Over the past 28 months the Mercury spacecraft, like all other flight vehicles, has had its share of weight growth. But fortunately, the spacecraft weight has not exceeded tolerable limits. This fact is not without reason, however. Considerable effort and attention have been given to devising the lightest possible systems compatible with the present state of the art.

A summary of the weights of the Mercury spacecraft is given in figure 12. An exploded sketch of the four major assemblies of the spacecraft with their respective weights is shown at the top of the figure. The escape tower assembly with the escape motor loaded weighs a total of 1,073 pounds. The antenna canister weighs 86 pounds and the retropackage with all six solid-propellant motors loaded weighs 262 pounds. The weight of the basic spacecraft, including the astronaut, is 2,524 pounds.

The total launch weight of the spacecraft, 3,945 pounds, is the sum of the weights of the four assemblies. The 4,124-pound gross launch weight shown in the figure includes the combined weight of the four assemblies as well as the weight of the launch-vehicle adapter and main clamping band. The orbit weight listed in the figure includes the weights of the retropackage, the basic spacecraft, and the antenna canister less the propellant of the posigrade separation motors. The reentry weight of 2,586 pounds is the combined weight of the basic spacecraft plus antenna canister, less 24 pounds of reaction control fuel and coolant water expended in orbit.

In order to give an idea of the weight allotment to the various systems onboard the Mercury spacecraft, a further breakdown of the basic system weights of the orbit configuration is given. The structural weight, which includes 60 pounds of insulation, but not the heat shield weight, is less than one-fourth of the total orbit weight. The weight given for the stabilization and control system includes that of both the automatic and manual systems as well as the weight of the reaction control system and its 55 pounds of hydrogen peroxide fuel. The landing and recovery system weight includes all parachutes, the landing impact bag, and the various recovery aids. The next weight item includes the cabin instrumentation and displays, cameras, and the periscope. The electrical group includes such items as batteries, inverters, wiring, and so forth. In addition to the radio communications, command receivers, and radar tracking beacons, the communications system includes the telemetry and onboard recording system. The environmental control system has a total weight of 132 pounds. The next item includes the weight of the astronaut, his 24-pound pressure suit, and the survival pack. The last item of 22 pounds accounts for manufacturing variations between the various production spacecraft. This figure is slightly less than 1 percent of the total orbital weight of 2,866 pounds.

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SPACECRAFT AND ESCAPE-SYSTEM CONFIGURATION

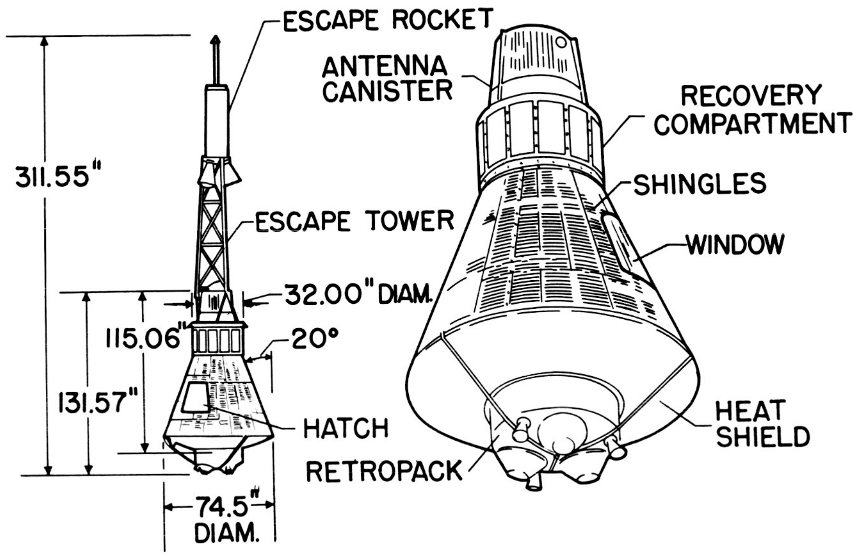


Figure 1

SPACECRAFT IN WEIGHT AND BALANCE FIXTURE

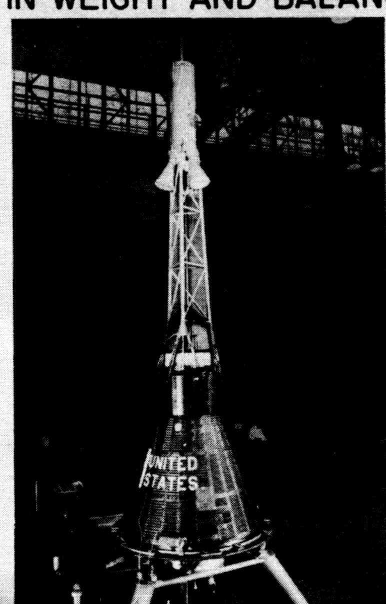


Figure 2

S-61-55

[redacted]

SPACECRAFT ENTRANCE HATCH AND OBSERVATION WINDOW

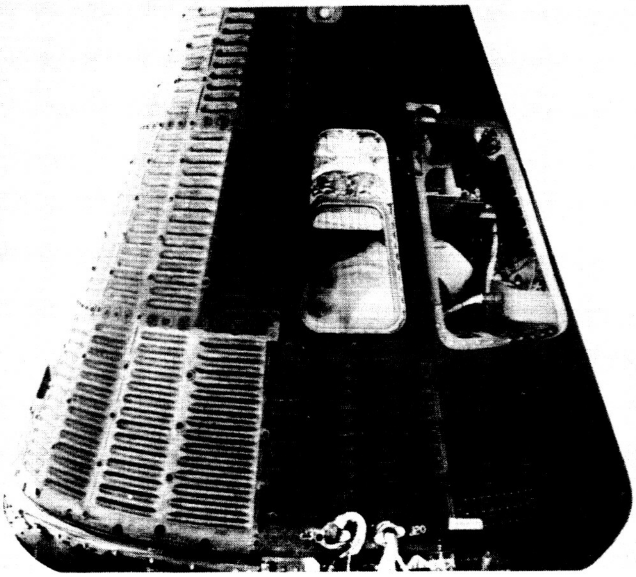


Figure 3 S-61-56

SPACECRAFT STRUCTURE

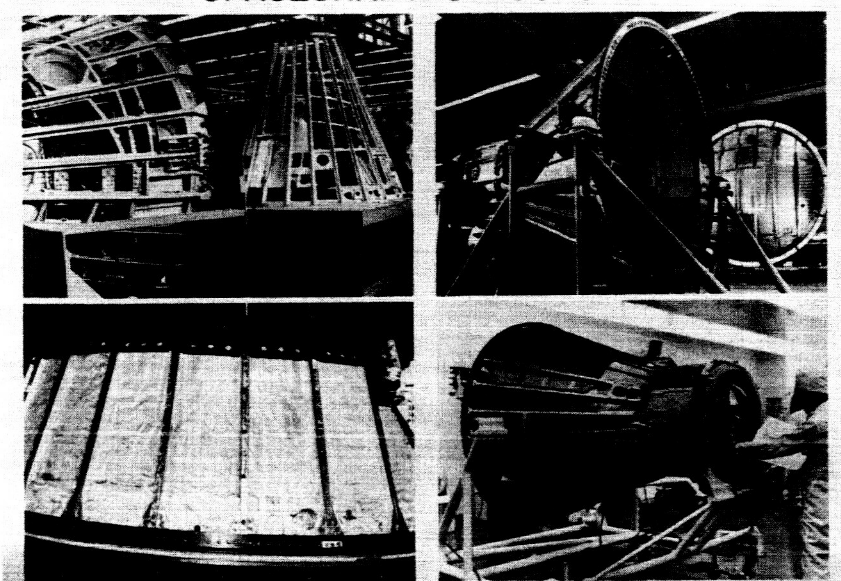


Figure 4 S-61-37

[redacted]

SPACECRAFT HEAT PROTECTION

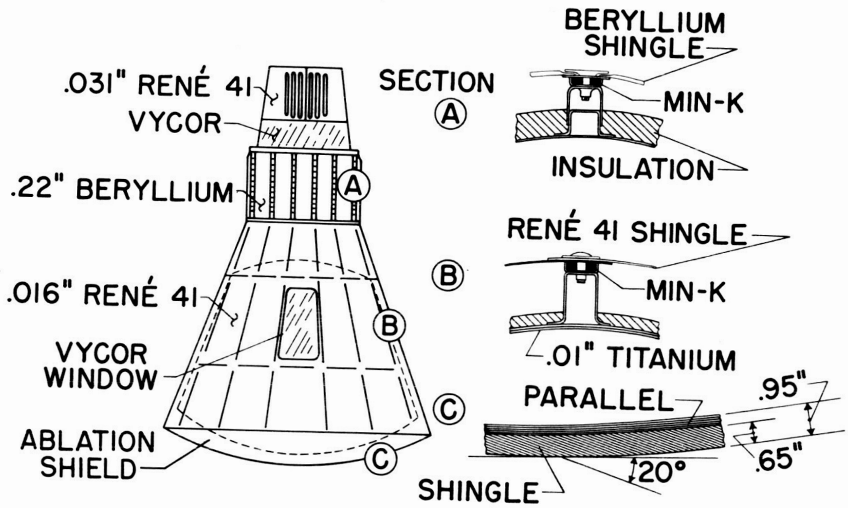


Figure 5

SPACECRAFT INTERNAL ARRANGEMENT

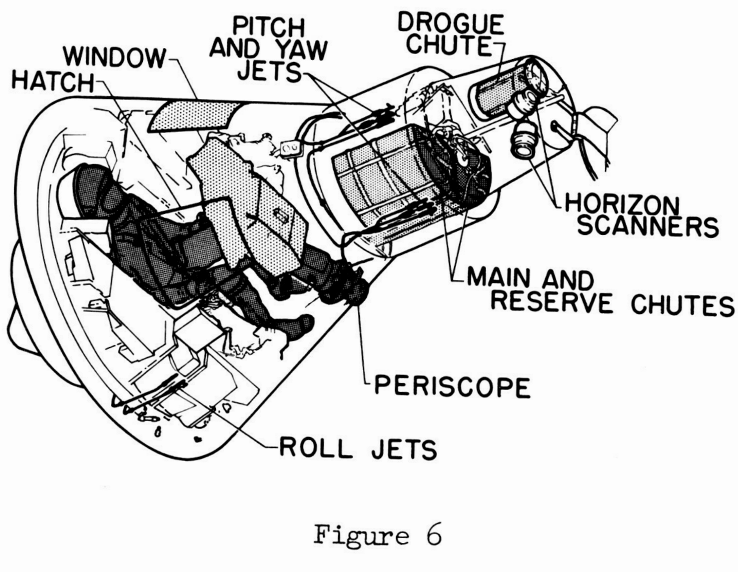


Figure 6

THREE-AXIS HAND CONTROLLER

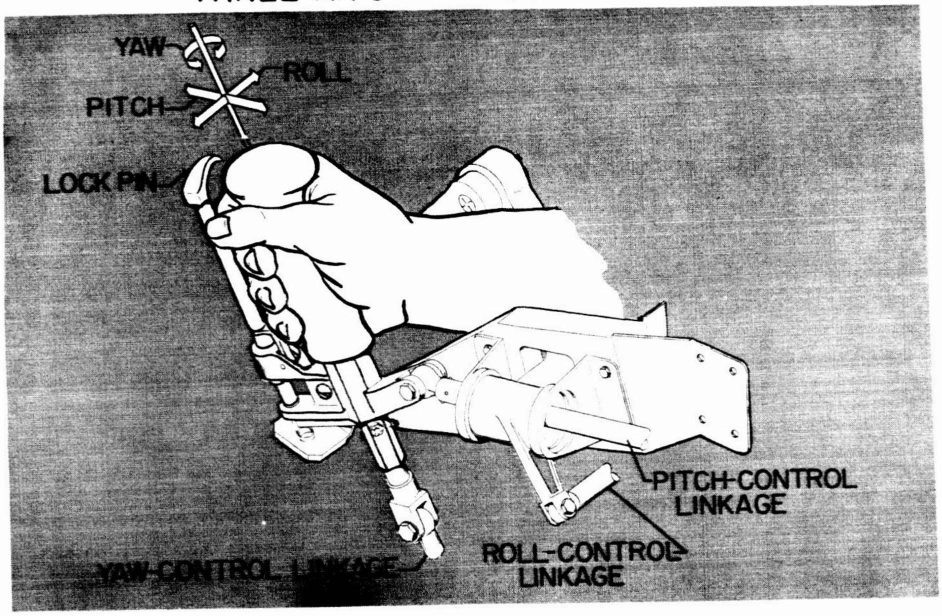


Figure 7

ASCS SEQUENCES—NORMAL MISSION PROFILE

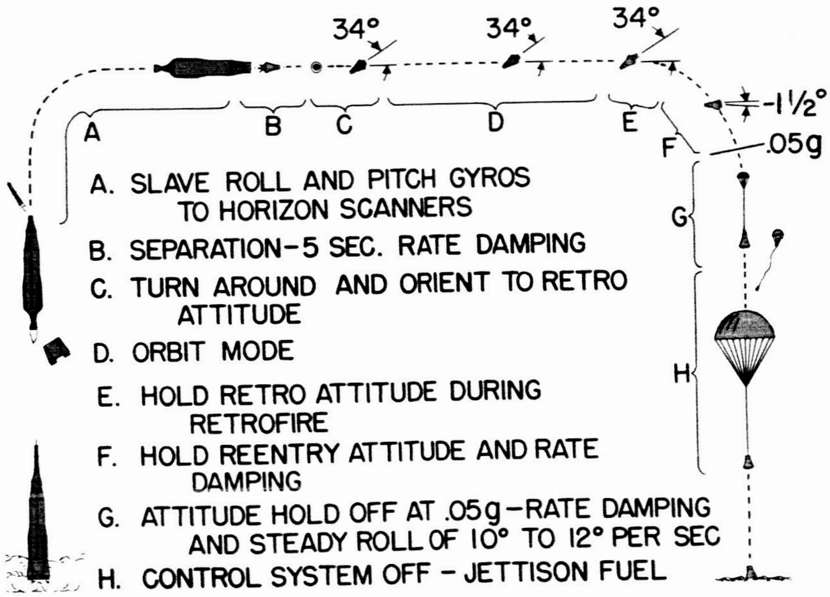


Figure 8

[redacted]

SPACECRAFT INSTRUMENT PANEL

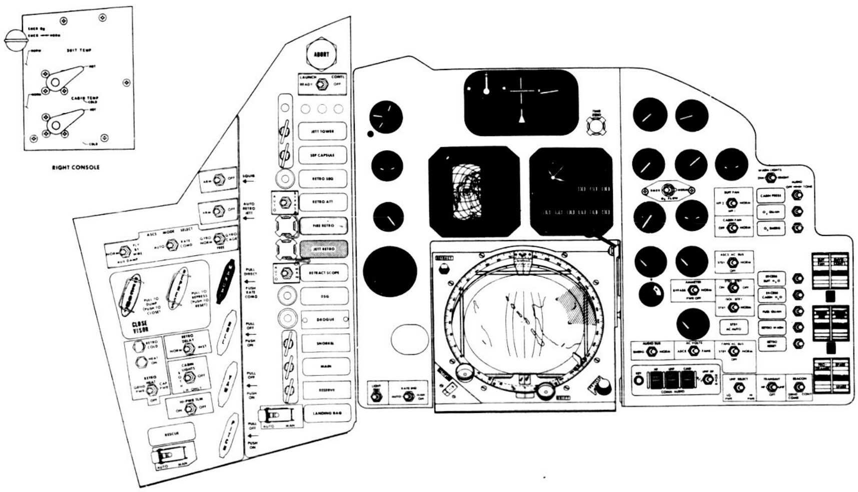


Figure 9

IMPACT ATTENUATION

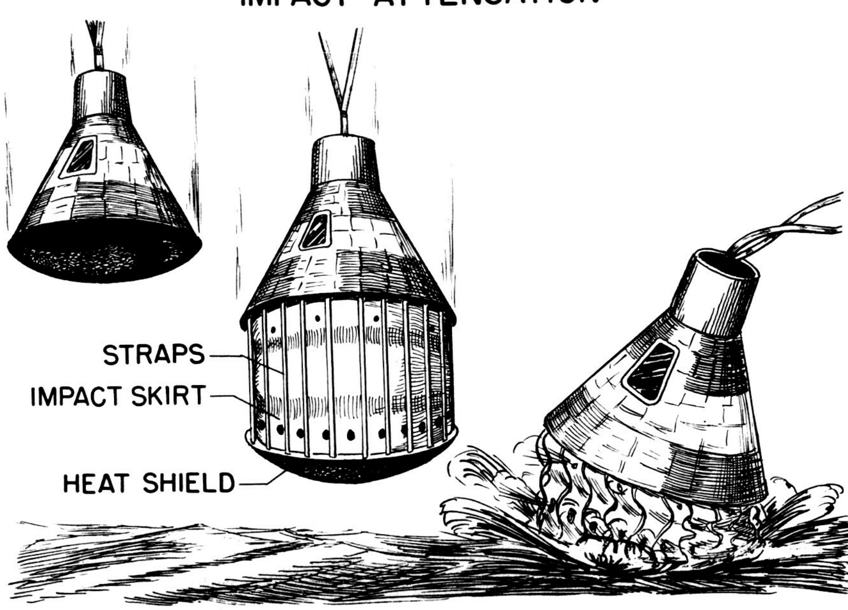


Figure 10

[redacted]

LANDING IMPACT BAG INSTALLATION

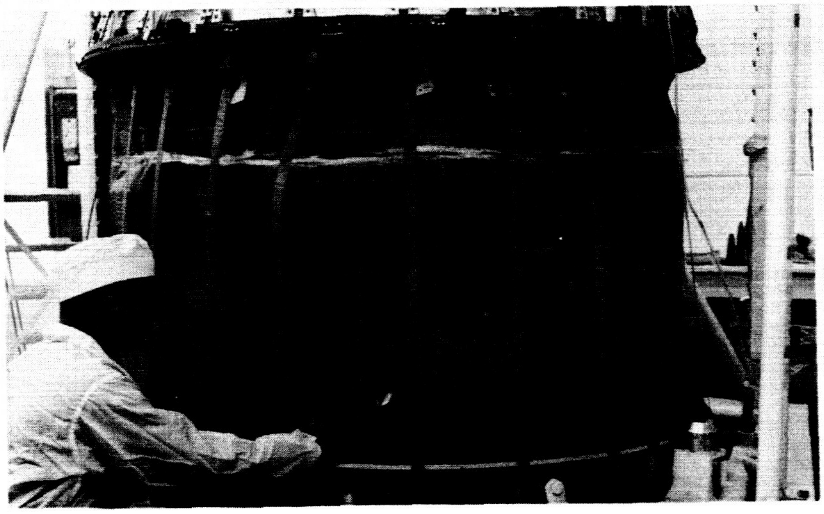
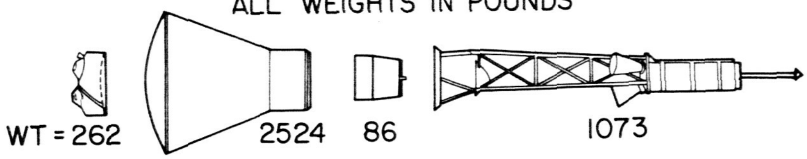


Figure 11 S-61-57

SPACECRAFT WEIGHT SUMMARY
ALL WEIGHTS IN POUNDS



ORBIT-CONFIGURATION WEIGHTS	
STRUCTURE	619
HEAT SHIELD	316
STABILIZATION AND CONTROL	272
RETROSYSTEM	281
LANDING AND RECOVERY	348
CABIN EQUIPMENT	107
ELECTRICAL SYSTEM	315
COMMUNICATIONS	208
ENVIRONMENTAL CONTROL	132
CREW AND SURVIVAL EQUIP.	246
MANUFACTURING VARIATION	22
TOTAL	2866

	WT
GROSS LAUNCH	4124
ORBIT	2866
REENTRY	2586

Figure 12

[redacted]

35

MERCURY SPACECRAFT STRUCTURAL DEVELOPMENT PROBLEMS

By Caldwell C. Johnson, Jr., and Leslie G. St. Leger

NASA Space Task Group

INTRODUCTION

When the contract for Project Mercury was let in February 1959, it was given a high national priority and an intensive manufacturing schedule was arranged. In order to meet this delivery schedule a policy was established by the National Aeronautics and Space Administration that detail engineering, research and development, and manufacture of the spacecraft would proceed simultaneously. It was known that this was a questionable and unprecedented course to take for a project of this nature, but this action was felt justified by the urgency of the program.

There are two important facets to the successful implementation of simultaneous design, development, and manufacture. First, a comprehensive set of design criteria which include all normal and single malfunction emergency modes of flight must be established at the start of the program. Second, production control must be made flexible so that engineering changes in design which are inevitable in such a plan can be easily incorporated.

In the case of Project Mercury, a set of loading conditions was established during the early stages of design, and a philosophy of catering for single system failures was included. Double system failures were considered outside the scope of the design. This philosophy resulted, generally speaking, in an efficient and reliable structure. However, because of the lack of experimental data in some cases, and changing requirements in others, some structural modifications became necessary later on in order to cater for design conditions that were not initially included. It should be pointed out that these critical structural design conditions do not occur in the exotic space flight regime, but rather in the mundane and brief periods during atmospheric launch and landing, particularly during an emergency mode of operation.

The major loading cases which influence the Mercury structure during exit are as follows:

Maximum αq of 7,500 deg-lb/sq ft
Abort at any altitude
Booster explosion
Control engines "hard over"

[redacted]

During reentry, the major loading cases are:

- Maximum deceleration and heating
- Drogue parachute deployment
- Antenna fairing jettison
- Main or reserve parachute deployment
- Water or land impact

An ultimate factor of safety of 1.5 was applied to all loads.

STRUCTURAL DEVELOPMENT PROBLEMS

A few examples of structural development problems and the consequent modifications necessary are shown in figures 1 to 7. In almost every case, the engineering solution to the problem was straightforward, but in some instances the necessity for the engineering modification was difficult to establish in view of the hardware production status. These examples are presented not so much to point at the specific problems, but rather to illustrate the importance of a careful preliminary study of design criteria before actual fabrication gets underway. All the figures have a similar format. A circle on the sketch indicates the general structural area being discussed. On the figure is listed the nature of the problem, whether it is a normal or emergency event, the basic reason for the problem, and, lastly, the remedy.

Escape Tower Heating

The heating problem associated with the escape tower is summarized in figure 1. Initial estimates of the aerodynamic heating of the escape tower during exit indicated that only moderate levels of heating were to be expected on the tower elements, and hence a conventional material, SAE 4130 steel, was selected as being adequate. However, subsequent more refined calculations pointed up the fact that there existed certain areas of very intensive localized heating, and that temperatures of the horizontal members and at the joints of the open truss structure would reach as high as 1,600° F during a normal flight. This oversight was a direct outcome of not emphasizing all the thermal design criteria at the outset of the program. The remedy in this case was to add some fiber-glass insulation to the horizontal members, and no production bottlenecks were created.

Tumbling Abort at Maximum Dynamic Pressure

Tumbling abort at maximum dynamic pressure (fig. 2) was not considered initially because some early wind-tunnel tests conducted by NASA indicated there was adequate static stability at all expected angles of attack. These tests, however, because of simulation difficulties, did not include the effects of the escape-rocket jets. During the evolution of the design, the escape vehicle configuration was changed somewhat so that the center of gravity moved further aft and the center of pressure moved further forward; therefore, static stability no longer existed at all angles of attack. This fact, together with the eccentricity of the escape-motor thrust vector which is required for a satisfactory "miss distance" from the launch vehicle, resulted in the realization that tumbling of the spacecraft during an abort was possible and should be catered for in the design. The effect on the aerodynamic loads was approximately to double the bending moment at the base of the tower, and to increase greatly the external pressure on the afterbody of the spacecraft. The remedy from a manufacturing point of view was not welcome. Existing escape towers were scrapped and redesigned to cater for the increased loads. The fittings that formed the joint between the cylindrical and conical portions of the spacecraft were redesigned in steel to replace the existing titanium parts. The factor of safety at the clamp ring of the tower to spacecraft joint for this condition was reduced below the required 1.5, but was approved on the basis of the low probability of the event occurring. The effects of the increase in external pressure were fortunately easy to remedy. This problem was counteracted by effectively increasing the cabin internal pressure during atmospheric flight by substituting a pressure relief valve operating on a differential system rather than the absolute system used previously. In the original design, the cabin pressure was vented to atmosphere from sea level up to an altitude of 25,000 feet, whereas the present design holds the cabin internal pressure at 5.5 psi above ambient throughout the launch phase of flight. The differential collapsing pressure acting on the structure was thereby reduced to tolerable proportions during a tumbling abort.

Afterbody Shingles

The shingles on the afterbody are designed for operation at relatively high temperatures, since they must dissipate the reentry heating by radiation. In addition to the thermal environment, the shingles must remain structurally intact in an environment of considerable noise. (See fig. 3.) There was little experience on the flutter behavior of the curved corrugated panels at high temperature, and a development test program prior to the design of hardware would have been very beneficial. The shingles became suspect after the reduction of temperature data from the Big Joe flight in September 1959, which showed higher-than-expected temperatures on the shingles surrounding the antenna fairing and the recovery compartment. To cater for these higher heat inputs, the shingles on the antenna fairing were replaced by shingles of increased thickness made of the superalloy, René 41. The heat-protection system for the recovery compartment was redesigned by utilizing 0.22-inch-thick beryllium panels. After the Big Joe flight, an intensive qualification testing program by NASA and the McDonnell Aircraft Corp. was undertaken at the Daingerfield Facility and at the Langley Research Center to check, at high temperature, the flutter behavior of the 0.010-inch-thick cobalt-alloy shingles on the conical section of the spacecraft. Although the results of these tests were not conclusive because of exact simulation difficulties, sufficient doubt was cast on the integrity of the original cobalt-alloy shingles for a decision to be made by Space Task Group management to replace them with shingles of similar corrugated design, but with the material changed to René 41 and the thickness increased from 0.010 inch to 0.016 inch.

Drogue-Parachute Deployment

Drogue-parachute deployment is a normal sequential event that occurs at 21,000 feet at the start of the landing phase of flight, but the design condition stems from an emergency mode associated with the failure of the automatic stability and control system and means that the drogue parachute may be deployed at any angle to the spacecraft axis. An early postulate that the rotational response of the spacecraft would reduce the opening load for this condition proved to be wrong, and the original calculated opening load of 1,500 pounds proved to be underestimated when compared with later experimental results. This early postulate resulted in an antenna fairing structure that was understrength and that had to be reworked to resist an opening load of 2,250 pounds acting at any angle to the spacecraft axis. (See fig. 4.) Much of the trouble would not have existed had an early research and development program been established to rationalize opening parachute loads.

Antenna Ejection

The antenna fairing is ejected at an altitude of 10,000 feet by the firing of a mortar. The recoil loads from the mortar were obtained from simulated firing tests early in the program, but unfortunately the results proved to be on the low side because of improper simulation of the structural dynamics of the support structure. (See fig. 5.) This improper simulation, together with the fact that the redesign of the antenna fairing had resulted in an increased weight, caused the failure of some fasteners and a main structural web in the cylindrical section during static firing tests. The weakness was remedied by increasing the size of fasteners and the thickness of web. The remedy in this case was simple both from engineering and manufacturing points of view.

Reserve-Parachute Deployment

The reasons for the structural problems during the reserve-parachute deployment (fig. 6) are much the same as for drogue parachute deployment. In the event of a main-parachute failure, the reserve parachute can be deployed at any angle to the axis of the spacecraft. This fact, coupled with the increased loads due to the growth in weight of the spacecraft, resulted in low factors of safety on the support structure. Some strengthening has been carried out and some of the low margins of safety have been accepted because of the extremely low probability and the implications from a manufacturing viewpoint. Weight growth seems to be an inevitable feature of all designs and should be borne in mind from the onset of a program.

Water or Land Impact

The original concept for Project Mercury was a water landing, with impact on land considered a not very likely emergency mode. Later rational analysis of escape trajectories "off the pad" revealed that a land impact was a distinct possibility and should be included as a design condition. (See fig. 7.) For a water landing, the original specification was for a vertical descent rate of 30 feet per second, with a maximum surface wind of 18 knots. A subsequent review of the official weather data of the North Atlantic Ocean revealed that unless operations were to be restricted to certain times, a more realistic design criteria would be a surface wind velocity of 30 knots and waves equivalent to sea state 4. The original crushable honeycomb structure beneath the astronaut was not adequate to mitigate the deceleration forces or onset rates from these extended design conditions, and an air bag suspended beneath the spacecraft was devised. The impact skirt has proved an efficient device for attenuating deceleration forces and has also been instrumental in improving the flotation stability of the spacecraft during the postlanding period.

CONCLUDING REMARKS

The examples given have highlighted the structural development problems of Project Mercury. These problems have now all been satisfactorily resolved, and there is high confidence in the reliability of the structure to fulfill all the missions for which it is designed. The examples have been given in order to emphasize the importance of carefully surveying at the beginning of a program all possible design conditions that may be encountered during all phases of the flight, from prelaunch to postlanding. The early crystallization of comprehensive structural design criteria that include growth possibilities and a flexible production control can substantially reduce subsequent development problems.

ESCAPE TOWER HEATING

- AERODYNAMIC HEATING DURING LAUNCH CALCULATED TOO HIGH FOR TOWER STRUCTURE
- NOMINAL FLIGHT MODE
- EARLY TOWER HEATING ESTIMATES TOO LOW
- INSULATION ADDED TO TOWER MEMBERS

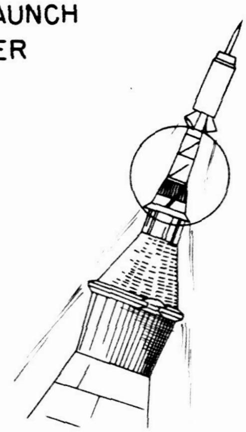


Figure 1

TUMBLING ABORT NEAR q_{MAX}

- COMBINED EFFECT OF MOTOR THRUST AND BODY BENDING CALCULATED TO OVERSTRESS TOWER AND SUPPORTING STRUCTURE
- EMERGENCY FLIGHT MODE
- STABILITY AND LOAD FACTOR ANALYSES NOT UP-DATED AS CONFIGURATION UNDERWENT ENGINEERING CHANGES
- ESCAPE TOWER REBUILT, SPACECRAFT STRUCTURE LOCALLY REINFORCED

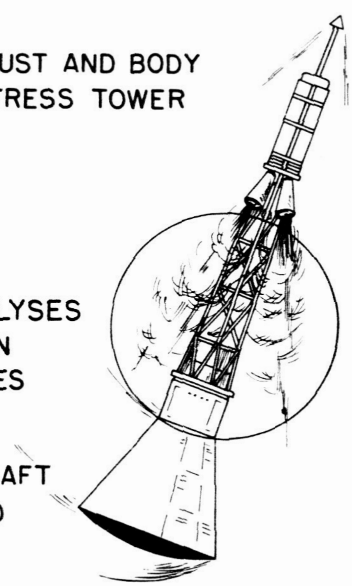


Figure 2

AFTERBODY SHINGLES AT REENTRY

- COMBINED EFFECTS OF NOISE, FLUTTER, AND AERODYNAMIC HEATING COULD CAUSE SHINGLE FAILURE
- NOMINAL FLIGHT MODE
- LOCAL FLOW CONDITIONS REASSESSED; SUBSEQUENT WIND-TUNNEL DATA EVALUATED AND BIG JOE DATA APPLIED
- SHINGLES REMADE OF NEW AND THICKER MATERIAL

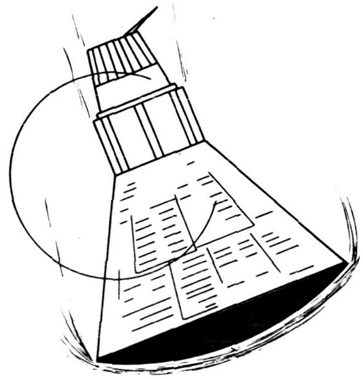


Figure 3

SPACECRAFT AT DROGUE DEPLOYMENT

- HIGH PULLOFF ANGLE AT DEPLOYMENT WOULD OVERSTRESS SPACECRAFT STRUCTURE
- EMERGENCY FLIGHT MODE
- DESIGN REQUIREMENTS NOT RATIONALLY ESTABLISHED
- ANTENNA FAIRING STRENGTHENED

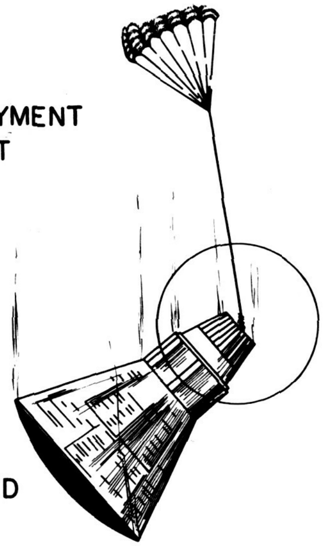


Figure 4

ANTENNA EJECTION

- MORTAR RECOIL OVERSTRESSED STRUCTURE DURING EARLY GROUND TESTS
- NOMINAL FLIGHT MODE
- PRELIMINARY EXPERIMENTAL PROGRAM DID NOT ADEQUATELY PREDICT MORTAR RECOIL

  ANTENNA FAIRING WEIGHT GROWTH FURTHER INCREASED RECOIL FORCE
- SPACECRAFT INTERNAL STRUCTURE REWORKED AND STRENGTHENED

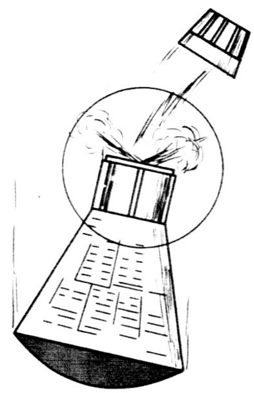


Figure 5

RESERVE PARACHUTE DEPLOYMENT

- RESERVE PARACHUTE OPENING FORCE APPLIED AT HIGH PULLOFF ANGLES WOULD OVERSTRESS STRUCTURE
- EMERGENCY FLIGHT MODE
- DESIGN REQUIREMENTS WERE NOT RATIONALLY DEVELOPED

  SPACECRAFT WEIGHT GROWTH FURTHER AGGRAVATED PROBLEM
- STRUCTURE REWORKED

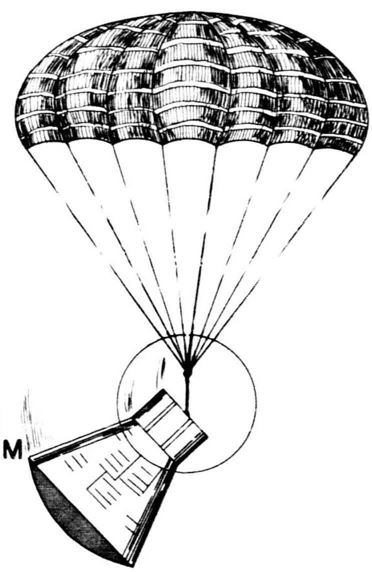


Figure 6

LANDING IMPACT ON WATER OR LAND

- HORIZONTAL DRIFT AND OSCILLATION COMBINED WITH VERTICAL DESCENT VELOCITY WOULD OVERSTRESS STRUCTURE AND CREW

- NORMAL LANDING MODE FOR WATER; EMERGENCY LANDING MODE FOR LAND

- COMPREHENSIVE SURVEY OF SURFACE WINDS AND OFF-THE-PAD ESCAPE TRAJECTORIES NOT INCLUDED IN ORIGINAL DESIGN CRITERIA

- IMPACT SKIRT ADDED

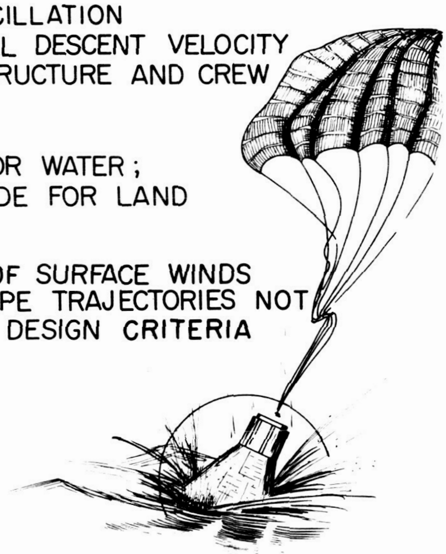


Figure 7

DESIGN FOR OPERATION

By John D. Hodge, John J. Williams,
and Walter J. Kapryan

NASA Space Task Group

INTRODUCTION

In this paper are discussed the operational requirements that influenced the design of the Mercury spacecraft. Many of these requirements are obvious and are taken into account in the normal course of design. Some are less obvious and should be emphasized in order not to be overlooked in the design of the Apollo spacecraft.

It should be pointed out that the original requirements for the Mercury spacecraft were for a simple design using basically known and well used components and that, although the outline of an operational concept was available, the majority of detail considerations for operations were not studied until after the general configuration was established.

With these thoughts in mind, it is well to define what is meant by operations. This is an all-encompassing word involving the history of the vehicle and supporting systems from delivery through recovery and the subsequent analysis for future action. The areas of concern in operations are 36

(1) Prelaunch check-out, from both a vehicle and a support-system viewpoint

(2) Launch operations

(3) In-flight operations

(4) Recovery and return

(5) Transportation

(6) Sequence system

As can be seen, these areas involve a myriad of disciplines and many agencies - private, federal, military, and international - each with its own special requirements.

DISCUSSION

Prelaunch Check-Out

Prelaunch check-out probably has the most effect on the design of the spacecraft. Careful attention to design will allow less time in preparation and will result in less time lost when changes are necessary. Some of the factors involved in this area are discussed in the following sections.

Accessibility and modular design.- Space and weight limitations prevented as much attention to this as was desirable. It is important to be able to remove equipment that needs servicing or replacement as quickly as possible. Particular items of concern here are batteries, power supplies in general, environmental control system components such as CO_2 absorbers, and instrumentation packages. Also, where items are difficult to remove, there is always the attendant risk of damage. An interior view of the Mercury spacecraft is presented in figure 1; the limited space in which to work on the spacecraft is shown in figure 2.

Flexibility of instrumentation.- The program inevitably involves variations in instrumentation from time to time, particularly after launches where problems have occurred. Flexibility is important here. Particularly the design must include the capability of patching into the system at various pertinent points without disturbing the basic operation of the system. It should not be necessary to undo connectors in order to patch into check-out equipment. Figure 3 shows a typical cable hookup for a spacecraft simulated flight test.

Functional check-out.- Wherever possible functional check-out is desired; this involves putting the spacecraft through a normal sequence of events with end-to-end measurements and as few artificial stimuli as possible. For the Mercury spacecraft, an altitude chamber is used for the environmental control system check-out and a two-axis, controlled-rate, gimbal rig is used for the automatic control system check-out.

Range interference.- One thing to remember with regard to testing at Cape Canaveral Missile Test Annex is that any test involving radiation of radio frequency (RF) requires range clearance and, therefore, this kind of test should be avoided if possible. Of course, the RF systems themselves require radiative tests, but the use of the RF system - for example, telemetry - to check out other spacecraft systems should be avoided.

Ground-support equipment.- With regard to ground-support equipment in general, it is most important to tailor the check-out to a research and development program and to the type of personnel involved. It must be remembered that highly trained engineers and technicians are available for this task and that this allows the use of simple equipment which, in turn, allows more flexibility of testing. Some automatic or semiautomatic equipment is desirable, particularly in the area of circuit checks where the deduction process that man can apply is not required. There is some question whether it is either wise or necessary to exchange engineers or technicians who are knowledgeable on the vehicle and capable of check-out with simple equipment for computing and ground-support-equipment engineers and technicians who usually have to struggle to keep their equipment functioning. Ground equipment referred to herein is that for check-out and problem diagnosis. In-flight vehicle check-out, with a small crew and limited space, poses a different problem and one which requires much study particularly when considering a lunar landing and take-off.

For the Mercury spacecraft, check-out trailers were used. These trailers were chosen to allow for a multiple-site usage. For example, the same equipment was used at NASA Wallops Station, George C. Marshall Space Flight Center, and Cape Canaveral. However, this does mean that work space is cramped and extreme care must be taken when the trailers are moved and set up at another site. Figure 4 shows an interior of a spacecraft check-out trailer.

Launch Operations

Requirements of launch operations must be considered in the design of a spacecraft. Several factors in this area are pertinent to design and are discussed separately.

Launch complex compatibility.- Several gantry modifications were required. These modifications included the addition of work platforms and the necessary power supplies and equipment stands. In addition, the total work area was enclosed in what is essentially a white room (fig. 5) to avoid contamination of spacecraft systems. The umbilical system provided no problems with the Redstone, but on the Atlas pad there was an incompatibility between the position of the umbilical plug on the spacecraft and the umbilical tower; this resulted in a somewhat tortuous path for the cable. (See fig. 6.) A dual separation system, mechanical and electrical, was used, but because of the problem mentioned redesign of the mechanical backup system was necessary to ensure reliability of operation. It seems that, no matter how many electrical and gaseous lines are available in the plug, there are never enough after a few flights have been made. Some spare booster lines were used for some blockhouse functions for the Mercury spacecraft. Once again, it must be remembered that radiation requires range clearance and the use of hardline testing can certainly save time.

Spacecraft—launch-vehicle interface.- In the area of spacecraft—launch-vehicle interface, it is essential that simple mechanical attachments be provided. Imagine the problems associated with an awkward bolt or connector 250 feet above the pad. The Mercury adapter is fitted to the launch vehicle prior to attachment of the spacecraft. The spacecraft-adapter joint uses a simple Marman type band with three explosive bolts. Adequate clearance between the spacecraft and launch vehicle is essential to allow access into this region and it has been found that access hatches are necessary in the adapter to avoid "blind" fitting. Initially, these hatches were not available in the Mercury-Atlas adapter and it was necessary for the first fit to be made with a man inside the adapter. (See fig. 7.)

Electrical interface is limited to a few wires associated with the automatic abort system and with lift-off signals; however, overall vehicle testing simulating the complete sequence of events was considered mandatory in order to validate these connections.

It is well, at this point, to mention the use that is made of solid-state electronic devices. The trend to this type of equipment is based on added reliability. However, experience gained from Project Mercury has shown that there is a large susceptibility to power transients; in many cases, malfunctions or even failure of the equipment resulted. These transients are present in any circuitry associated with switching and the firing of pyrotechnics, and the solution has been to isolate individual systems by using diodes or filters, or to provide separate power supplies where necessary.

Access to spacecraft.- Simple access to the spacecraft is essential if work is to be carried out without delays and damage to the systems, and this applies in the hangar as well as on the pad. Hatches must be large and work areas adequate. These were restricted for the Mercury spacecraft because of necessary limitations in size; as a result, pad check-out is more time consuming than is desirable. Hatches and work areas adequate for normal crew use may be completely inadequate for vehicle preparation.

Emergency egress.- If an emergency arises on the pad, the procedure for egress from the spacecraft is to blow the explosive bolts around the hatch. This procedure has proved feasible, although undesirable. Mechanical hatches are preferable; however, weight limitations precluded their use for the Mercury spacecraft. A cherry-picker is used to assist in the rapid descent of the astronaut from the spacecraft should this be required. (See fig. 8.) It is unlikely that this simple approach is feasible for vehicles much higher than the Redstone and the Atlas.

Pyrotechnic arming.- In the area of pyrotechnics, the two major considerations are safety and time to arm. Much thought was given to the use of blast deflectors under the escape tower; in fact, these are provided on the Redstone gantry. The final procedure was to remove all shorting plugs prior to entry of the astronaut. Figure 9 shows four of the shorting plugs in the area of the retropack. About 20 of these plugs are required for the Mercury spacecraft, each of which must be individually grounded; and this is a time-consuming procedure.

It must be remembered that the igniter circuits are not armed at this time and the closure of a series of switches is necessary to achieve this. Attention to detail design increases the safety and reduces the total time to perform this procedure. In particular, the shorting plugs should be as close to the igniter as possible and should be simple to install and remove.

Perhaps the point to be made from this discussion of launch operations is that procedures should be as simple as possible and that the basic vehicle should be disturbed as little as possible by the check-out; by this means the total countdown time can be reduced and the safety of the operation can be improved.

Launch-delay tolerance.- Probably the maximum delay allowable after entry of the astronaut is dependent on the ability of the astronaut to stay within the spacecraft. This delay may be no more than about 2 hours for a Mercury-Atlas mission, including the normal time required for check-out. Another delay is associated with the requirement for daylight recovery. The Apollo system will be limited by the available length of the parking orbit and the position of the moon at the time of launch. However, it is important that the vehicle be designed to be able to make use of this available tolerance. This has been done for the Mercury spacecraft for all times except probably after umbilical cable drop when holds are limited to about 30 minutes because of cooling difficulties.

In-Flight Operations

In-flight operations for the Mercury spacecraft are so designed that the astronaut may participate extensively in system management and spacecraft control. However, satisfactory mission control from the ground relies on the passage of a given amount of data between the spacecraft and the network stations. The procedures associated with this operation are discussed in a subsequent paper by Christopher C. Kraft, Jr., and C. Frederick Matthews. Some of the equipment problems that these procedures have imposed on the design of spacecraft are discussed. The areas of concern are

(1) Voice communications

(2) Telemetry transmission and associated instrumentation

(3) Command reception

(4) Tracking beacons

(5) Antenna patterns

Again, because of the limitations of power and cooling capability, it was necessary to establish duty cycles for this equipment. After a detailed operational analysis in this area, additional power and cooling were made available in order to ensure a continuous passage of data between the ground and the air while the spacecraft was within range of a given tracking station. For a manned mission the primary method of communication is voice. For this purpose a dual ultra-high frequency (UHF) system is used with a high frequency (HF) backup system.

Telemetry transmission is available and is the primary source of data for an unmanned mission. For the Mercury spacecraft dual telemetry transmitters were provided for redundancy, and parameters which were required for real time monitoring were made available on each link. Once again flexibility is desirable in this area. Both continuous and commutated channels are used, and for some parameters commutated segments are strapped to provide a greater data rate.

For command reception, dual receivers are provided in the spacecraft. An operational input was to provide for adequate security on this circuit and multitone systems were used to prevent inadvertent command reception.

Tracking beacons were found to need relatively quick recovery times in order to take into account the overlapping capability of the Mercury network where several radars would be interrogating the beacon during the same time period.

Antenna patterns received considerable attention operationally. Because of the shape of the Mercury spacecraft, the large blunt face effectively shields radiations forward. It was found necessary to orient the spacecraft in a higher nose-up attitude while in orbit than originally had been planned in order to ensure adequate reception of signals by the tracking stations.

The Bermuda tracking situation shortly after sustainer-engine cutoff gave considerable concern. Modifications to the antenna system were proposed in order to facilitate continuous tracking during the turnaround maneuver and to allow the computer to complete its go—no-go computations. The point to be raised is that compatibility with the range facilities and operational procedures will sometimes compromise the design of the spacecraft, particularly when the use of existing facilities is specified.

Recovery and Return

The operational concept for recovery and the procedures which are followed are discussed in a subsequent paper by Robert F. Thompson, William C. Hayes, Jr., and Donald C. Cheatham. In addition to the normal impact point which is obtained from tracking and the computing system, search and location of the spacecraft is carried out by aircraft using electronic and visual techniques. Early tests of the recovery systems established the advantages to be gained by larger range capability in the beacons and the aircraft receivers; this capability was provided.

The initial concept using a balloon antenna for the HF recovery system was discarded in favor of a whip antenna because of design difficulties associated with erecting the antenna in high winds and rain.

Considerable operational inputs were provided concerning the seaworthiness and water stability of the spacecraft in the ocean area. Much testing was accomplished and center-of-gravity limitations were imposed in order to ensure an adequate safety margin during astronaut egress. Another area concerned wind and sea state design conditions which were increased to allow a greater probability of launching on a given day.

Survival equipment has remained essentially unchanged since the initiation of the original design. However, a personnel parachute was added to provide for manual escape in the low probability that the tower could not be separated after an abort.

Transportation

Transportation gave few extreme problems for the Mercury spacecraft. The size of the spacecraft enabled the use of air transportation. Some difficulty was encountered in loading and unloading from the aircraft. (See fig. 10.) Special attention to detail design was required. Within the facility, simple dollies were used. Again, details were important. The original dolly design precluded adequate access to the spacecraft and maneuvering inside the hangar was difficult. Another dolly was designed to rectify this problem. (See fig. 11.)

This new dolly is used for transportation to the pad. The normal recovery operation involves helicopters; several design problems were encountered, particularly in the area of the pickup loop on the spacecraft itself and the helicopter which had a quick release device which needed some modification to ensure safety of operation. (See fig. 12.) The original design emphasized safety of the helicopter and a change of emphasis was required. One other problem was the rotation of the spacecraft while on the helicopter. Several cables were broken during early exercises with boiler-plate spacecraft before a suitable modification was established.

With regard to equipment transportation, large amounts of check-out equipment, particularly electronic equipment, were placed in trailers in order to facilitate multiple use. In the original delivery, long hauls over roads were necessary, which entailed the normal problems of vibration. The transfer was made with a slow road speed and low-pressure tires and, in addition, some pieces of equipment were specially packaged.

Sequence System

The problem of sequence systems is now briefly discussed.

Design requirements.- The design requirements are twofold and paradoxical. First, the system should ensure the occurrence of a given event at the correct time and sequence. This usually will result in parallel redundancy. Second, the system should prevent the occurrence of a given event out of sequence. This requires a series redundancy. The resulting system inevitably becomes complex, and with the large number of signal paths available the probability of an erroneous signal becomes very large.

The variety of electromechanical sensors, such as pressure switches, microswitches, and relays, leads to an extensive vibration problem.

These two areas have been the major causes of sequence-system failures in the Mercury spacecraft. The lessons to be learned are that (a) redundancy can be overdone to the extent that reliability is reduced, and (b) where electromechanical systems are used, an extensive examination of the vibration spectrum is required.

System check-out.- The sequence system has proved to be ideally suited to a semiautomatic type of check-out. It has been necessary during hangar checks to examine every possible path of signal to eliminate the possibility of sneak circuits. In addition, the physical actuation of the electromechanical devices is considered necessary during this phase of check-out.

When the complete vehicle is on the pad, a total sequence test of the launch-vehicle—spacecraft combination is carried out. Because of the exigencies of time, the various events are simulated.

This latter test is of particular importance since the total vehicle is more susceptible to sneak circuits than are the individual components.

CONCLUDING REMARKS

Some of the operational requirements which have influenced the design of the Mercury spacecraft have been discussed. For Project Mercury, the formulation and establishment of these requirements involved the efforts of a large number of agencies and personnel. A series of points of contact within each of these agencies was set up and a number of coordinators within NASA who are responsible for different phases of the operation were appointed. These points of contact formed the basis of a series of small working groups whose task was to delineate the problem areas and to make recommendations with regard to the solution of these problems. Where necessary, subgroups were formed to examine particular detail problems, and after the solution of these problems the subgroups were then disbanded. Working groups were set up in the areas of spacecraft design itself, spacecraft—launch-vehicle interface, launch operations, network operations, and recovery operations. This procedure has proved to be very effective, especially during the early stages of the design.

Some basic ground rules associated with operational requirements should be established in order to ensure that these inputs are both timely and adequate. First, it is important to consider a detailed operational concept prior to the detail design phase of the vehicle itself. A testing philosophy should be determined during the early design stages. This philosophy includes preacceptance testing at the plant prior to shipment and should also include prelaunch testing conducted at the launch site. For Project Mercury the concept of minimal testing at the launch site has not been realistic. Preflight check-out has usually evolved to include detailed testing of all systems. As a result a paradox exists that so often occurs in operations. Every effort must be made to reduce the total time at the launch site but, nevertheless, design of check-out equipment must take into account the high probability that detail systems tests will be required. Ground support equipment necessary to implement preflight-test philosophy should be incorporated into design planning in the early design stages, and the development of ground support equipment and vehicle hardware should be concurrent.

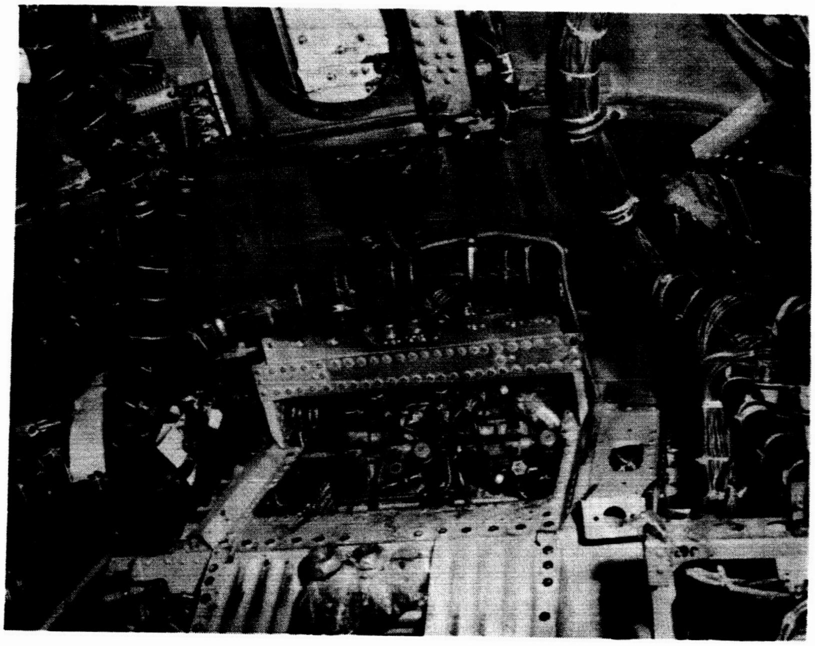


Figure 1 60-1213

Figure 2 LOD-61-6143

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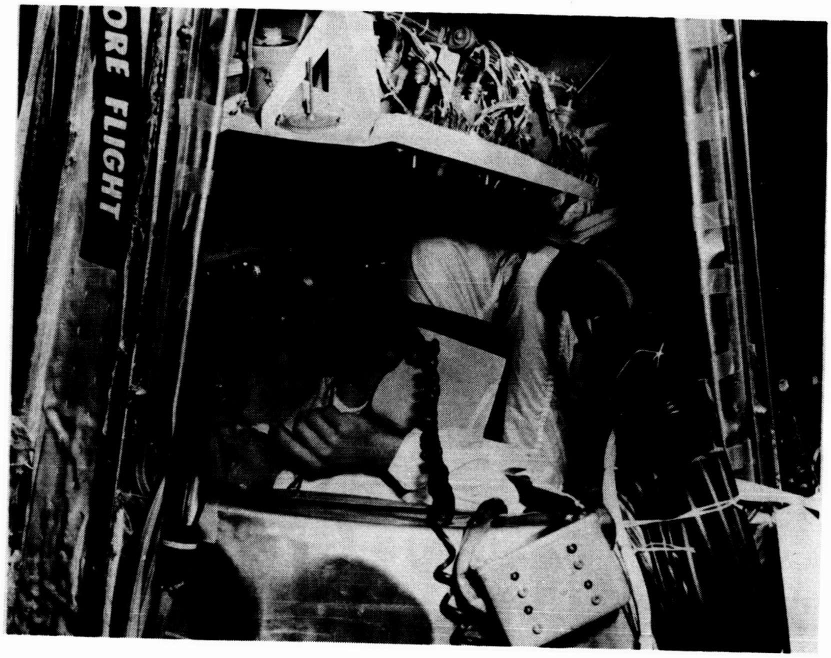

Figure 3 LOD-61-733

Figure 4 LOD-61-5952

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Figure 5 LOD-61-3776

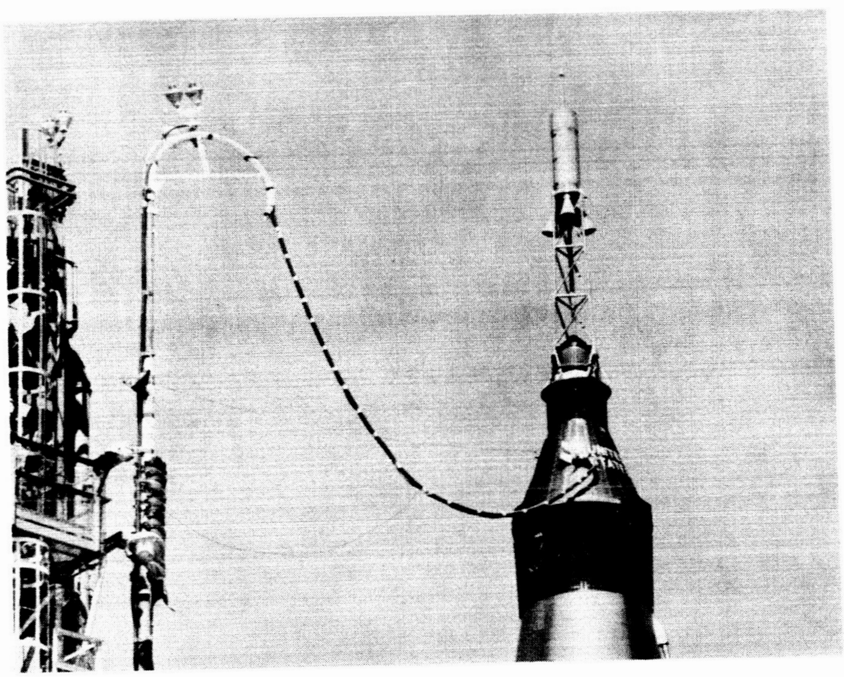

Figure 6 LOD-61-4141

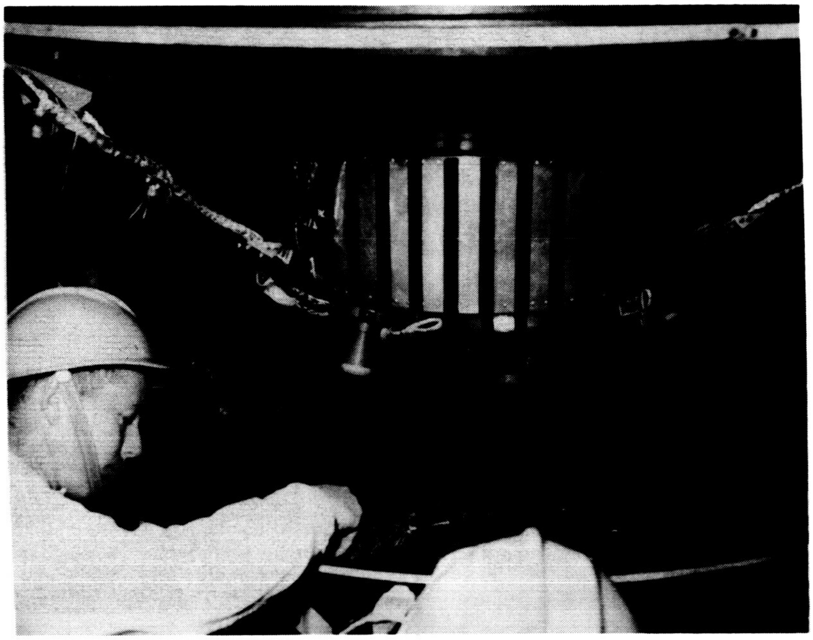

Figure 7 PL-60-59004

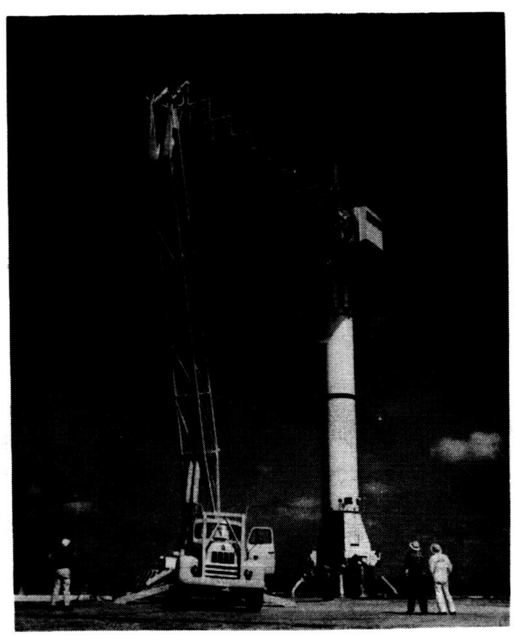

Figure 8 LOD-61-4405

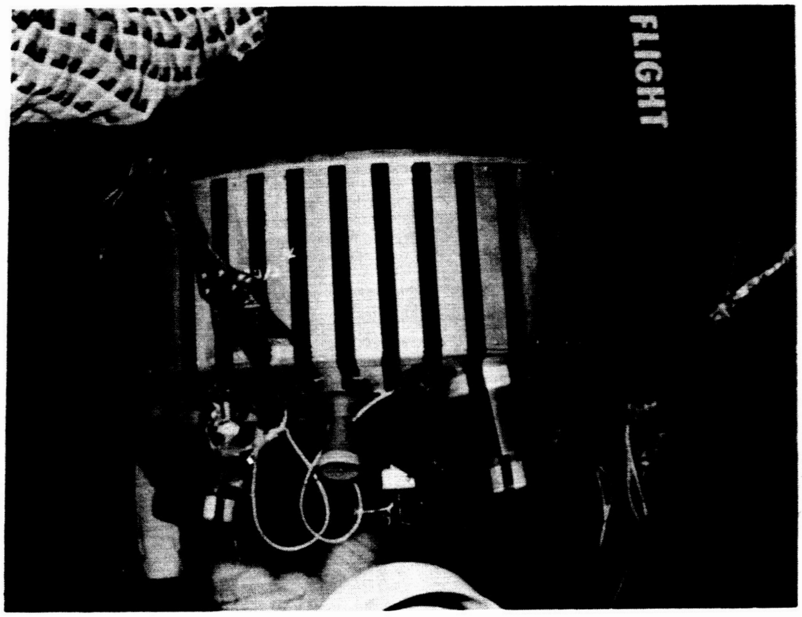

Figure 9 S-61-52

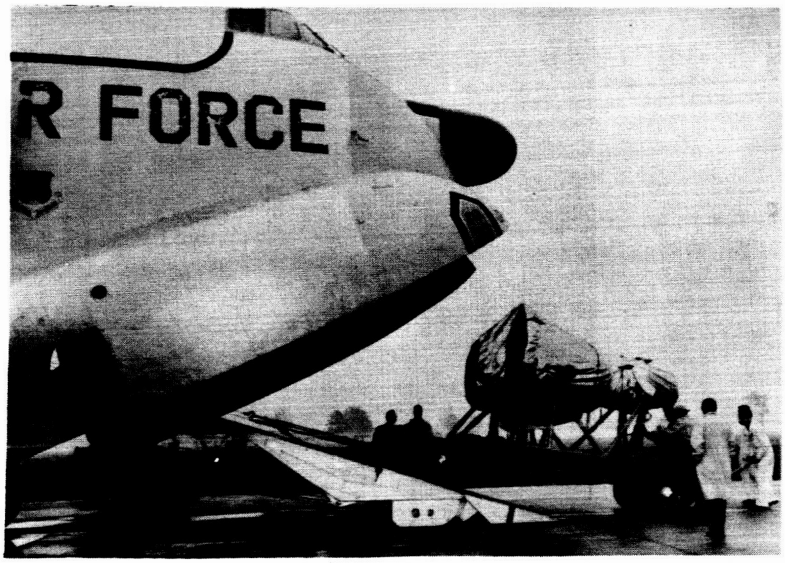

Figure 10 B-60-1666

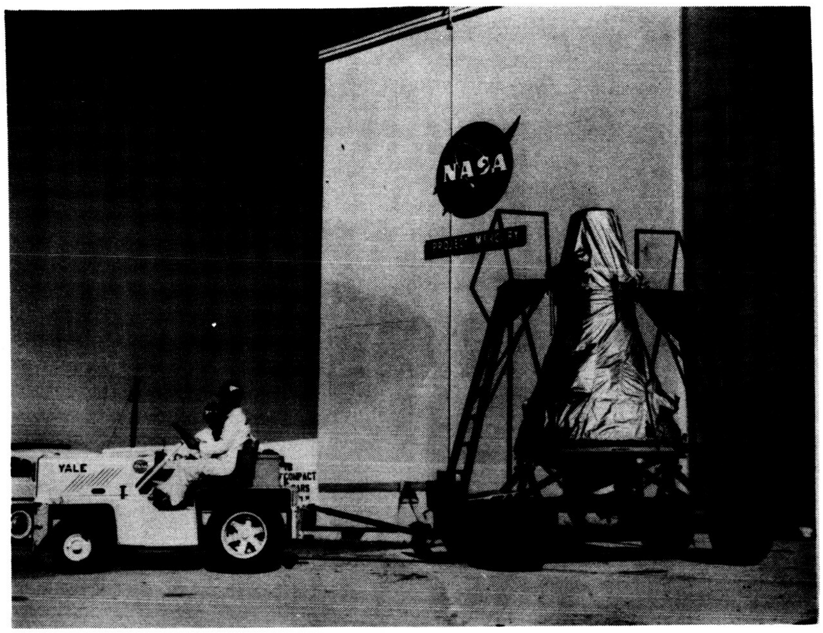

Figure 11 LOD-61-6013

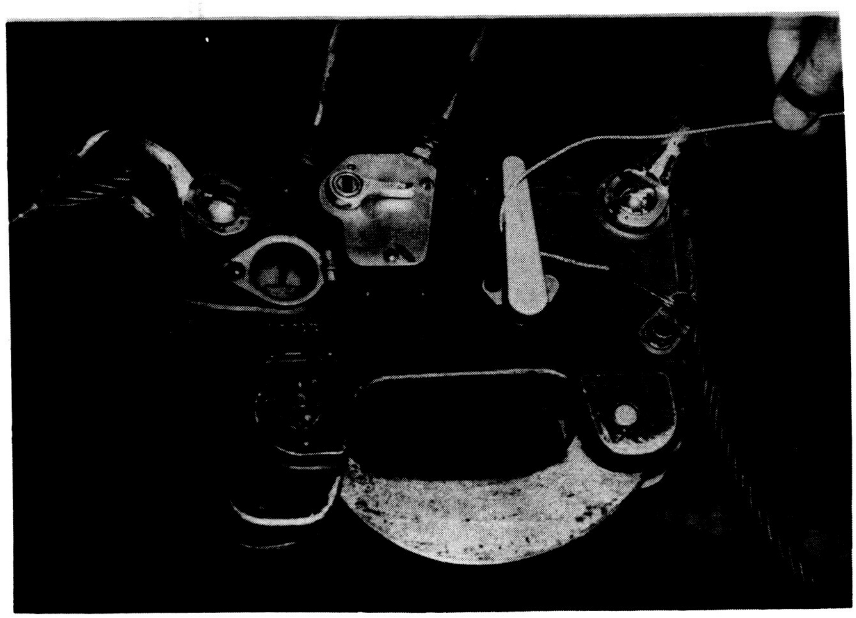

Figure 12 S-61-1499

MERCURY GROUND AND FLIGHT-TEST PROGRAM

By André J. Meyer, Jr., William M. Bland, Jr.,
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NASA Space Task Group

INTRODUCTION

In planning new projects, it seems appropriate to examine current projects for techniques and procedures worth adopting, as well as to note those to be avoided. This paper relates knowledge gained from experience with Mercury that should be helpful in the development of a test program for Apollo. In order to be assured that the Mercury spacecraft would perform properly in the ultimate flights, a tremendous number of tests were made on systems, subsystems, and individual parts.

The purposes and objectives of the extensive Mercury test program are to -

(1) Gain an understanding of design loads and conditions.

(2) Evaluate strength of structural components.

(3) Establish adequacy of thermal protection provided.

(4) Confirm operational characteristics of various systems and units under load.

(5) Check compatibility between electrical gear and communications equipment. 37

(6) Qualify parts under all types of environmental conditions.

(7) Determine component reliability and mean life to failure.

These same objectives apply equally well to any major space project, and Project Apollo is no exception.

The test program for Project Mercury can be divided into three distinct but overlapping phases: the wind-tunnel tests, the ground tests, and the flight tests. The discussion of these three phases will give an indication of the careful planning and organizing required to insure success of the Apollo mission.

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WIND-TUNNEL PROGRAM

The wind-tunnel program was conducted to check the aerodynamic feasibility of the Mercury shape both alone and in combination with various launch vehicles. The six basic configurations shown in figure 1 were investigated. In the abort configuration, the spacecraft must be stable with the escape tower leading in flight. Without the tower (the exit mode), it is desirable that the spacecraft be unstable in this same direction of flight. In fact, tunnel tests showed the need to add a destabilization flap to the small end of the spacecraft for this reason. The reentry configuration had to be stable with the large-diameter, blunt face forward. The other three configurations are of the spacecraft in combination with the three launch vehicles - Little Joe, Redstone, and Atlas. For Apollo, many more configurations and combinations of the different modules and launch vehicles will have to be investigated.

A total of 70 models varying in size from 1 percent of full scale to full scale were studied in over 100 tunnel tests for an accumulation of 5,500 hours of operation. Figures 2 to 9 show a few of the models and tunnel facilities used for Mercury. Twenty-eight different tunnels located in various parts of the United States were utilized. These facilities, their locations and operating capabilities, and the test objectives are presented in table I. Briefly, the test conditions are as follows: Mach numbers varied from 0 to 21, angles of attack varied from 0° to 180°, and Reynolds numbers ranged up to 15×10^{6}. Figure 10 shows the Reynolds numbers plotted against Mach number. For comparison with the data points, the test envelope of conditions expected during the exit and reentry phases of flight are also plotted. Although the spectrum was reasonably well covered by test points, very few tests were made at high Mach numbers because of facility limitations. Apollo, when it attains escape velocity, will reach speeds that are 38 percent faster than the maximum speed attained by Mercury. These facts point out the need for ground facilities of some kind that are capable of exploring extremely high speeds to support future space programs.

GROUND-TEST PROGRAM

By far the greatest number of tests in the Mercury program were carried out in ground-test facilities other than wind tunnels. Just to cite a few representative examples, the small hydrogen peroxide thrust chambers that control the attitude of the spacecraft were fired 160,000 times to develop and establish reliability. (Notwithstanding this large number of tests, occasional troubles are still experienced with these units.) A total of about 50,000 individual tests were made on parts of the Mercury environmental control or breathing system and 70,000 tests were made on communications hardware. All parts, bits, and pieces were subjected to some testing such as the environmental qualification tests outlined as follows:

Atmosphere tests
- Humidity and rain
- Water immersion
- Salt-water spray
- Sand and dust
- Fungus
- Compatibility with pure gaseous oxygen
- Temperature (high and low)
- Reduced pressure (5 psia)

Operating-loads tests
- Vibration (0 to 2,000 cps and up to 10g)
- Acoustic noise (up to 150 db)
- 15g shock (unit must continue to function properly)
- 100g shock (unit must not break free from its mount, must not start electrical fires, etc.)
- Acceleration (-12g to 20g)
- Burst pressure (pressure vessels, lines, and fittings)

Performance tests
- 150-hour age
- 1,000-hour life
- Power consumption (must not exceed specified values)
- Low voltage (function properly at below normal voltage)
- Radio interference checks
- Low current - no fire (to avoid inadvertent firing of pyrotechnics)
- Preinstallation acceptance (receiving inspection and check-out)
- Capsule system tests (after installation of system in spacecraft)

Since these tests are of the conventional type and many are explained in detail in numerous military specifications, no further explanation is presented herein.

Aside from the environmental tests, many systems and areas received much more than the normal amount of ground testing. Typical examples of critical areas are as follows:

(1) Flight simulation of complete spacecraft

(2) Loading of spacecraft structure while subjected to reentry heating

(3) Parachute and landing systems

(4) Environmental control system

(5) Astronaut egress checks

(6) Pyrotechnic firing

(7) Vibration and noise

(8) Heat shields

(9) Radio noise

(10) Couch and astronaut restraint system

(11) Gaseous toxicity and oxygen compatibility

(12) Optics and abrasion of periscope and windows

This list would probably apply equally well to Apollo with some other critical areas added. The complexity of some of these important ground tests is illustrated by a series of photographs (figs. 11 to 17) which were taken of the test setups for the first seven examples of critical areas listed. The 14- by 14- by 35-foot altitude chamber capable of simulating conditions up to an altitude of 300,000 feet and containing a production spacecraft is shown in figure 11. Quartz heating lamps and cooling coils simulate exit and reentry heating and heat losses to outer space while orbiting. In this test setup, all onboard systems are operated and all sequences of events are performed at the correct time and condition.

In figure 12 (a photograph taken during the structures test), the spacecraft is almost totally obscured by the quartz heating lamp and whiffletree loading fixtures. At first, only one early spacecraft structure was specifically assigned to these structures tests. Later, the first flight unit (off-the-beach abort flight) was also alloted to these tests; however, additional assemblies for the purpose would have benefited the Mercury program by accelerating this important type of testing. Figure 13 shows the test rig at McDonnell Aircraft Corp. used for dropping full-scale spacecraft at various horizontal and vertical velocities and different angles of attack. The rig was very useful for studying tumbling characteristics and for testing the strength of landing-impact bags and heat shields during land or water impact.

A special nonflight capsule was built to evaluate the environmental control system, that is, the system for providing a suitable breathing atmosphere for the astronaut. This unit with a man onboard

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was positioned in an altitude chamber (fig. 14), and a number of simulated missions were performed for extended time periods. Still another special capsule was constructed to investigate the ability of the astronaut to climb out through either the side or the top hatch in an emergency after impact on the water. Figure 15 illustrates this operation in one of the tow tanks at Langley. The flotation accuracy of the special assembly was checked by using another early spacecraft after having completed its flight mission. The tank with its wave-making machine was further used in endurance tests of the landing-bag system when the flight indicated this to be a problem.

Figure 16 shows the setup for firing an escape rocket to study heating and abrasion effects on the tower structure, windows, and the outer skin. Also the effects of high-altitude ignition and thrust magnitudes were determined. Similarly, all other Mercury rockets (posigrades, retrogrades, tower-jettison rockets, and the 1-, 6-, and 24-pound hydrogen peroxide thrust chambers) were fired many times. Other pyrotechnics, such as the explosive side hatch, the antenna canister, pilot and drogue parachute mortars, the squib electrical disconnects and valves, and the explosive bolts, were likewise fired many times under a wide range of conditions.

In figure 17 is seen the arrangement for vibration testing the spacecraft and the Atlas adapter with all onboard systems functioning. This photograph was taken at Langley Research Center but a similar assembly, except oriented in a horizontal instead of vertical attitude, was tested at McDonnell Aircraft Corp. This same spacecraft with a man in the seat was also subjected to an acoustical noise field at the exit of the Langley 9- by 6-foot thermal structures tunnel at sound levels duplicating those actually measured in a launch vehicle flight.

There were many other interesting ground tests conducted, such as heat-shield ablation tests, drop tests with dummies and pigs in couches, and the explosive-hatch test under water, but they are too numerous to describe herein. The ground-test program for Apollo will be more elaborate and will require more tests to guarantee reliability.

FLIGHT-TEST PROGRAM

The flight-test program was started about one-half year before the Mercury contract was let and is still moving ahead strongly. Flight tests are used to confirm the aerodynamic performance of the spacecraft as developed through the wind-tunnel tests and the proper flight operation of spacecraft systems as checked by the ground tests. Up to the present time, 133 flight tests have been completed. This number includes the very early boilerplate capsules such as shown in

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figure 18(a), which were merely dumped out of cargo aircraft to check parachute operations and sequencing. These were followed by the more complex and refined spacecraft (see fig. 18(b)) flown with the Little Joe launch vehicles and the first Mercury-Atlas vehicle called Big Joe. Finally, the production spacecraft as photographed in figure 18(c) was flown first from the beach by means of its escape rocket and then boosted by Little Joe, Redstone, and Atlas launch vehicles. All the Mercury launch vehicles are shown in figure 19. Of the 133 flights, only four spacecraft were totally destroyed. Two of these were destroyed because of launch-vehicle failures in which the spacecraft did not contribute to the failure. This is an excellent record since even the very first development drops and flights are included.

Only full-scale units were used in the flight program to avoid such complications as interpreting data by using questionable scaling factors. Full-size spacecraft also permitted the installation and early testing of spacecraft systems as soon as they became available and further permitted putting primates in these assemblies to gain information on the effects of high accelerations and weightlessness on living creatures.

A summary table of the flight program is presented as table II. The flight program for Apollo will undoubtedly be more elaborate than the Mercury program because it will cover escape, lunar orbit, and eventually lunar landing in addition to the goal of Mercury, that of orbiting the earth. Considerable thought already has been given to Apollo flight tests. In fact, some later Mercury spacecraft will probably be flown primarily to test systems proposed for Apollo.

EXPERIENCES BEARING ON APOLLO

The preceding information very broadly covered the details of the Mercury test program. The remaining discussion will concentrate on the real lessons learned through experience that should result in a more efficient, complete, and beneficial test program to support Apollo. It already has been shown that any adequate program will involve a very large number of individual tests and that some of the test setups will be very complex. Because of these factors, the program must be started early and should get just as much attention and priority as the spacecraft design and production. Several of the first structures built and much of the other hardware should be assigned to ground tests so that required changes discovered through testing can be introduced into the flight spacecraft as early as possible. For components where the imposed loads depend on their weight and the weight of other components, adequate margins on the test loads should be planned to provide for the inevitable weight growths and the extension of the original flight conditions. On

Mercury, despite a concentrated effort to hold weight down, the total spacecraft weight increased about 1/2 percent per month.

Experience has shown that compatibility checks and radio-interference tests also should be conducted early in the program with all communication, electrical and electronic gear, and associated wiring mounted in their proper relative positions and all systems operating simultaneously. Interaction problems can thus be discovered before all the components are built. Then filters, diodes, and other modifications can be added in later units to cope with voltage spikes, momentary power dropouts, and radio noise caused by the switchings, latching of relays, and pyrotechnic firings. It would be wise to build the compatibility mock-up so that it could be vibrated, shock loaded, and subjected to acoustical noise to detect sensitive relays, poorly fastened wires, etc.

Furthermore, particularly for electrical components, it was learned that much time and effort could be saved by devising and building complete bench setups of each major system so that components of that system can be checked out individually and immediately upon receipt from the vendor merely by plugging into the system mock-up.

Failures encountered during testing should be analyzed and reported rapidly so as to devise and perfect hardware changes as soon as possible and thus retrofitting will be minimized. It is human nature to proceed to the next test and to leave the less-interesting task of reporting for fill-in work.

It should be reemphasized that the Apollo test program should be started early and that an ample amount of hardware should be made available for ground testing as well as flight testing.

TABLE I.- WIND TUNNELS USED FOR MERCURY

Facility	Mach number	Test objectives
Langley Research Center Langley Air Force Base, Va.		
12-foot low-speed tunnel	0.05	Static and dynamic stability
8-foot transonic tunnel	0.25 to 1.20	Static and dynamic stability; pressure distribution
Unitary plan wind tunnel	1.6 to 4.65	Static and dynamic stability; pressure distribution, vibration, and flutter; heat-transfer characteristics
4- by 4-foot supersonic pressure tunnel	2.01	Static stability
20-inch hypersonic tunnel	5.98	Static stability; pressure distribution
11-inch hypersonic tunnel	6.70 to 9.60	Static stability; pressure distribution; heat-transfer characteristics
2-foot low-density hypersonic tunnel	3.02 to 6.82	Static stability
300-mph 7- by 10-foot tunnel	0.10 to 0.16	Static stability
20-foot free-spinning tunnel	0.09	Dynamic stability of reentry configuration and drogue parachute
9- by 18-inch supersonic aeroelasticity tunnel	2.0 to 3.0	Dynamic stability
Full-scale tunnel	0.05 to 0.10	Static stability
9-inch supersonic tunnel	1.94 to 2.91	Dynamic stability
Internal aerodynamics laboratory	0.60 to 1.40	Base pressure measurements on Little Joe plus spacecraft
Gas dynamics laboratory	4.95	Heat-transfer characteristics
Ames Research Center Moffett Field, Calif.		
Unitary plan wind tunnel	1.5 to 3.5	Static stability; vibration and flutter
14-foot transonic tunnel	0.60 to 1.20	Static stability; pressure distribution
2- by 2-foot transonic tunnel	0.6 to 1.3	Static stability
1- by 3-foot supersonic tunnel	1.38 to 4.06	Static stability
Supersonic free-flight tunnel	2.51 to 14.37	Dynamic stability
2- by 2-inch shock tunnel	5.0	Pressure distribution of reentry configuration; heat-transfer characteristics
10- by 14-inch hypersonic tunnel	5.0 to 6.0	Pressure distribution of reentry configuration; heat-transfer characteristics
Pilot-gun facility	6.0	Pressure distribution of reentry configuration
Lewis Research Center Cleveland, Ohio		
1- by 1-foot supersonic tunnel	2.47 to 4.89	Heat-transfer characteristics
Arnold Engineering and Development Center Tallahoma, Tenn.		
16-foot transonic propulsion wind tunnel	0.15 to 1.5	Static stability
50-inch hypersonic tunnel	8.08	Static stability; pressure distribution; heat-transfer characteristics
50-inch Hot Shot-2	17 to 21	Static stability; pressure distribution; heat-transfer characteristics
Ordnance Aerophysics Laboratory Daingerfield, Texas		
Engine test facility	2.44	Flutter and aerodynamic noise on outer shingles
McDonnell Aircraft Corp. St. Louis, Mo.		
Low-speed tunnel	0.15 to 0.27	Static stability; pressure distribution

[redacted]

TABLE II.- FULL-SCALE TEST PROGRAM

Tests	Method	1958 1959 1960 1961
1. Early impact studies	Drop test	(15)*
2. Early parachute tests	Airdrop	(10)
3. Drogue parachute tests	Aircraft pod	(15)
4. Extensive parachute tests	Airdrop	(57)
5. Research and development, escape-system tests	Off-the-pad	(3)
6. Research and development, entry test	Rocket flight	(1)
7. Research and development, escape-system tests	Rocket flight	(4)
8. Qualification, escape-system test	Off-the-pad	(1)
9. Qualification, spacecraft structure test	Rocket flight	(1)
10. Qualification, escape-system test	Rocket flight	(3)
11. Qualification, Mercury-Redstone systems	Rocket flight	(4)
12. Qualification, landing-bag tests	Airdrop	(17)
13. Qualification, Mercury-Atlas systems	Rocket flight	(2)
14. Freedom 7	Rocket flight	
		TOTAL (133)

*Numbers in parenthesis indicate number of tests.

PROJECT MERCURY WIND-TUNNEL CONFIGURATIONS

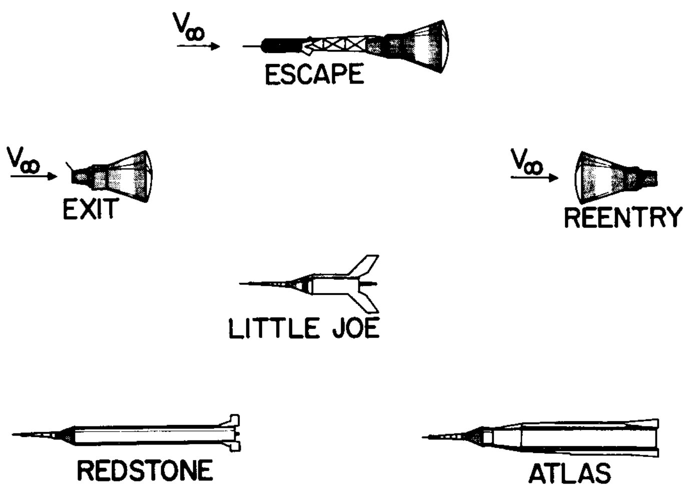


Figure 1

DECLASSIFIED

ABORT CONFIGURATION IN LANGLEY UNITARY PLAN WIND TUNNEL

Figure 2 S-61-72

MODEL OF SPACECRAFT ON LITTLE JOE LAUNCH VEHICLE IN LANGLEY UNITARY PLAN WIND TUNNEL

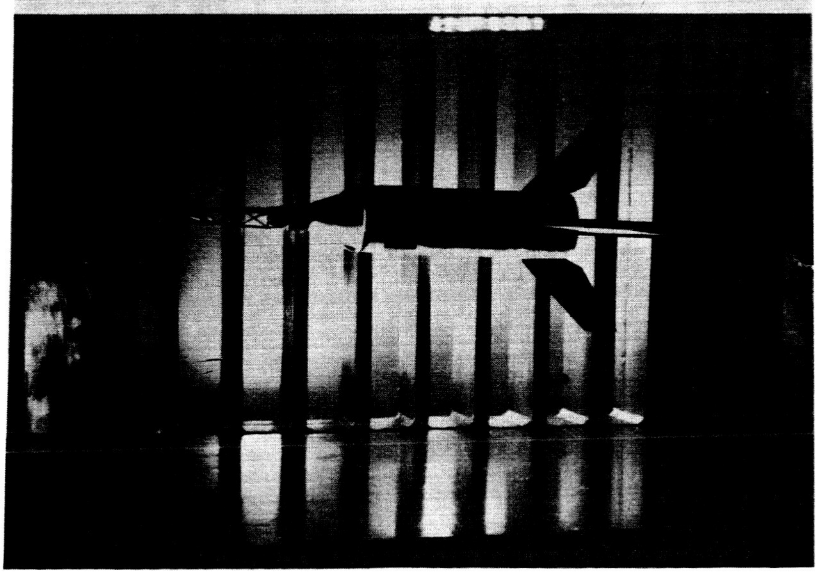


Figure 3 S-61-73

QUARTER-SCALE MODEL IN LANGLEY 12-FOOT LOW-SPEED TUNNEL

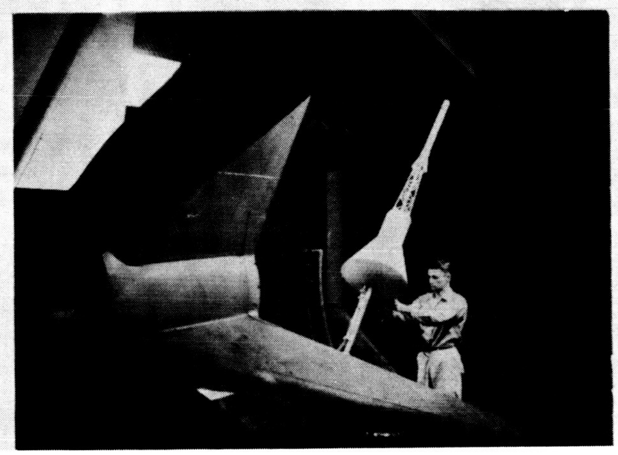


Figure 4 S-61-74

FULL-SCALE SPACECRAFT IN LANGLEY FULL-SCALE TUNNEL

Figure 5 S-61-75

[redacted]

SHADOWGRAPH OF 2.2-PERCENT MODEL IN AMES SUPERSONIC FREE-FLIGHT PRESSURIZED RANGE

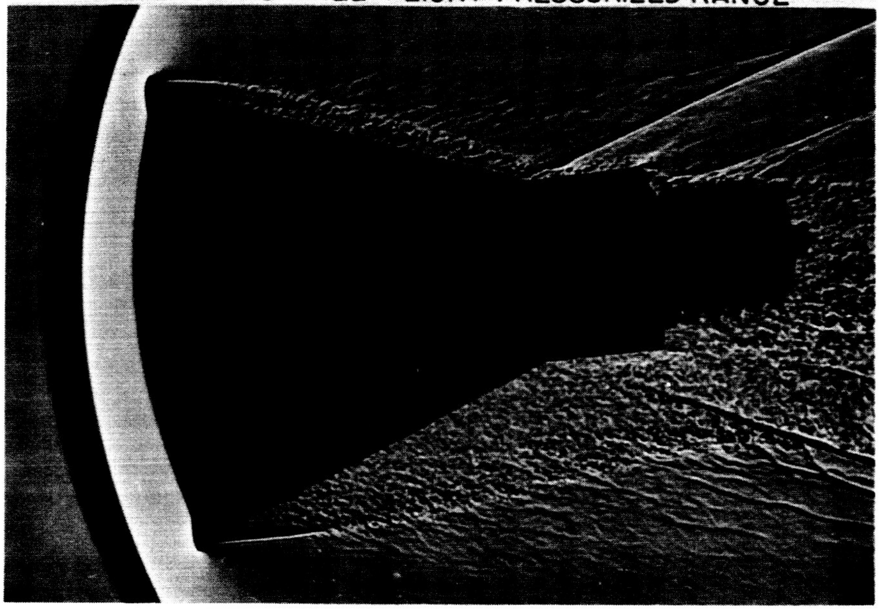


Figure 6

S-61-76

REENTRY CONFIGURATIONS IN LEWIS 1-BY 1-FOOT SUPERSONIC TUNNEL

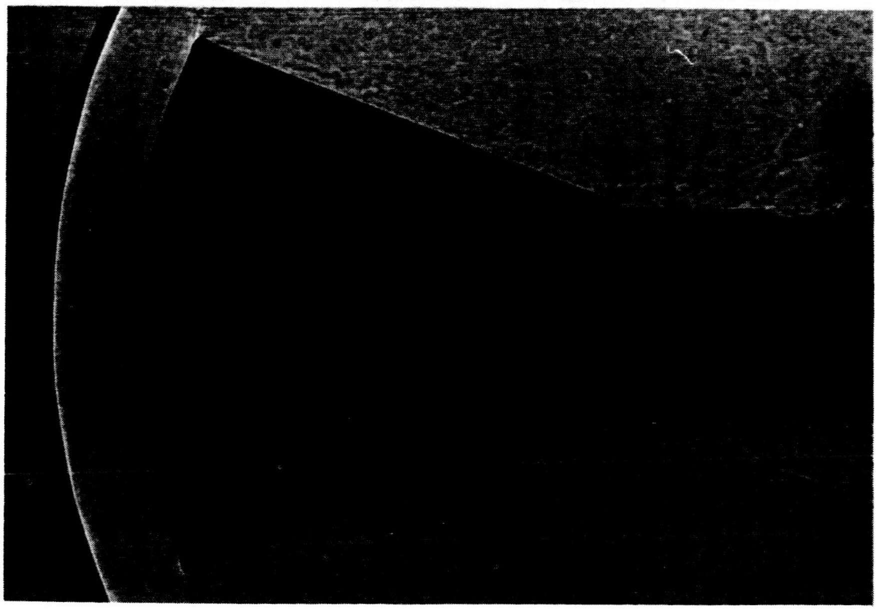


Figure 7

S-61-77

[redacted]

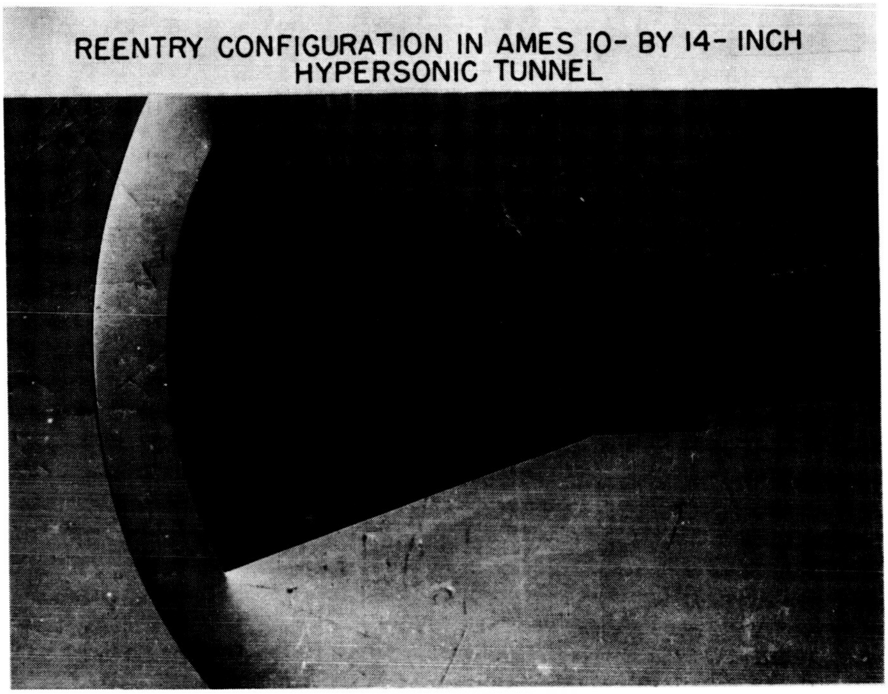


Figure 8 S-61-78

Figure 9 S-61-79

COMPARISON OF WIND-TUNNEL AND FLIGHT REYNOLDS NUMBERS

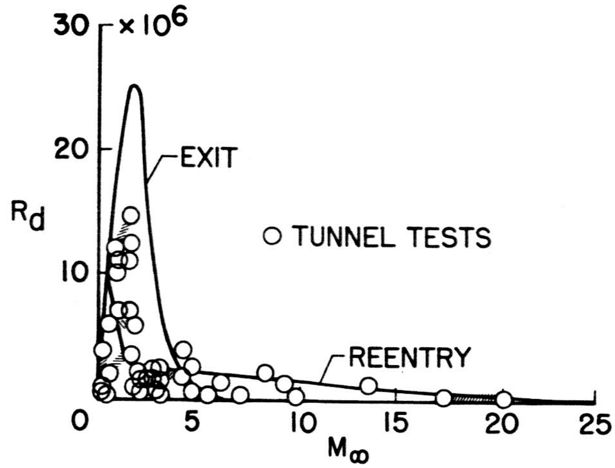


Figure 10

PRODUCTION SPACECRAFT IN McDONNELL ALTITUDE CHAMBER

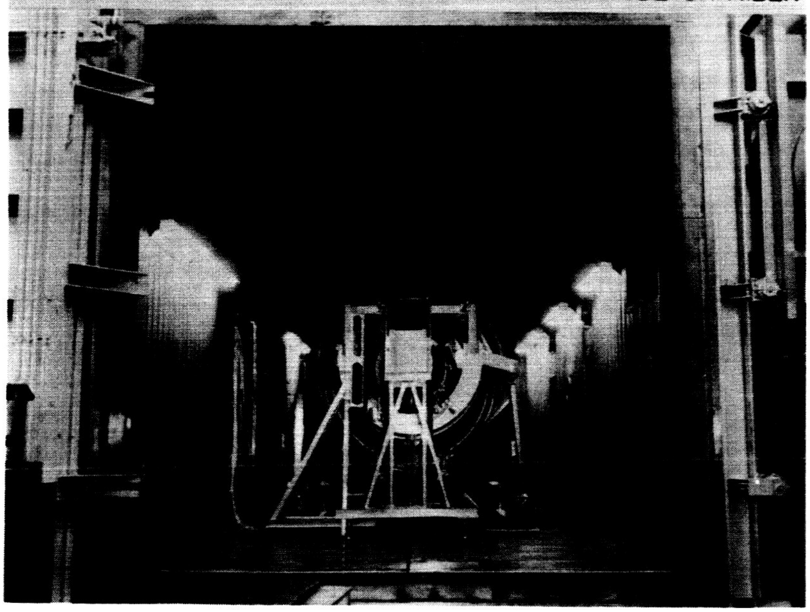


Figure 11 S-61-80

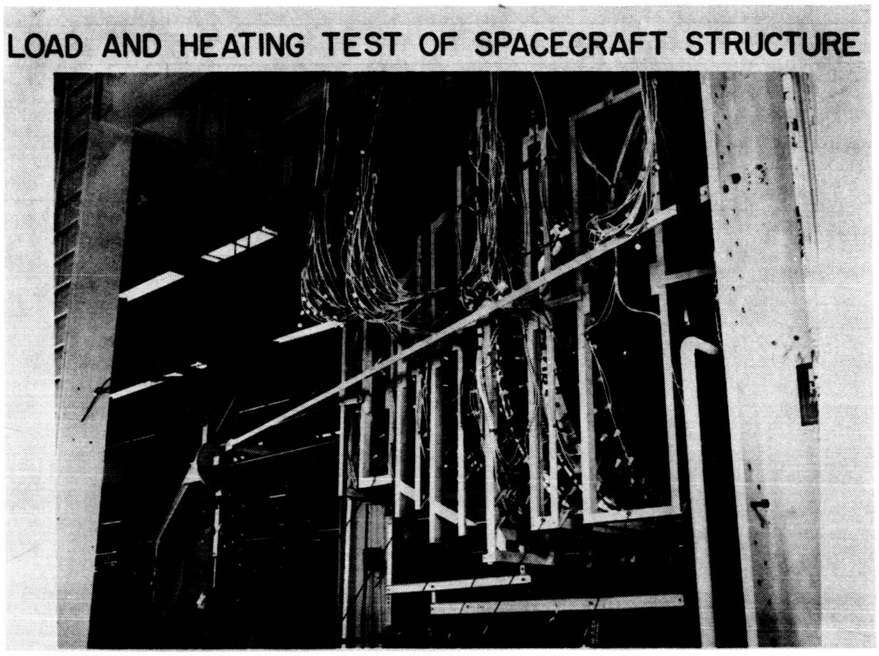


Figure 12 S-61-81

Figure 13 S-61-82

[redacted]

SPECIAL SPACECRAFT FOR TESTING ENVIRONMENT CONTROL SYSTEM IN AN ALTITUDE CHAMBER

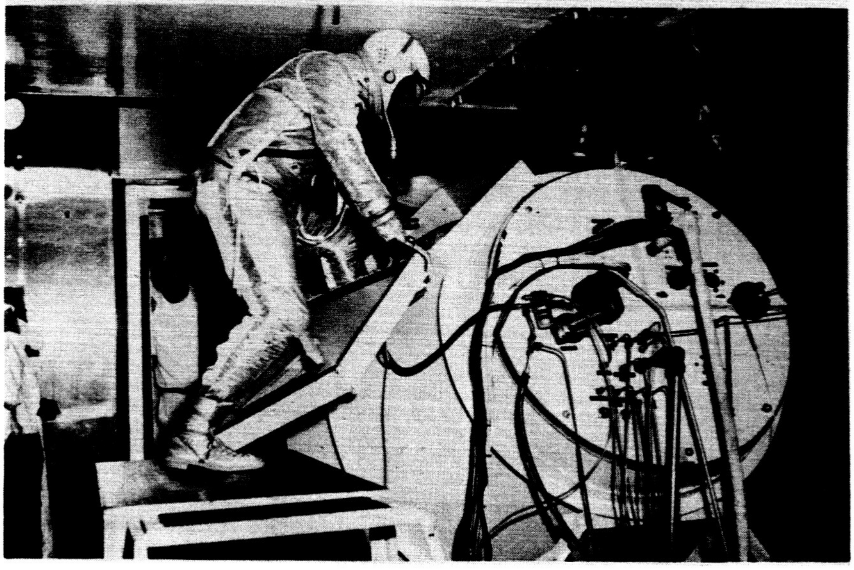


Figure 14 S-61-83

SPECIAL EGRESS TEST UNIT AND TRAINER IN LANGLEY TOW TANK

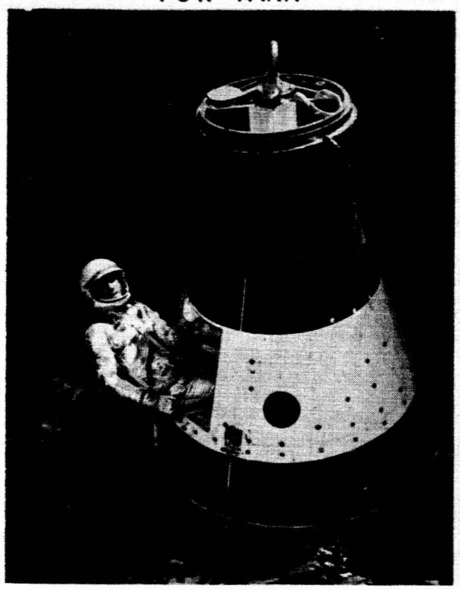


Figure 15 S-61-84

SETUP AT LEWIS RESEARCH CENTER FOR TESTING ESCAPE ROCKETS UNDER HIGH-ALTITUDE CONDITIONS

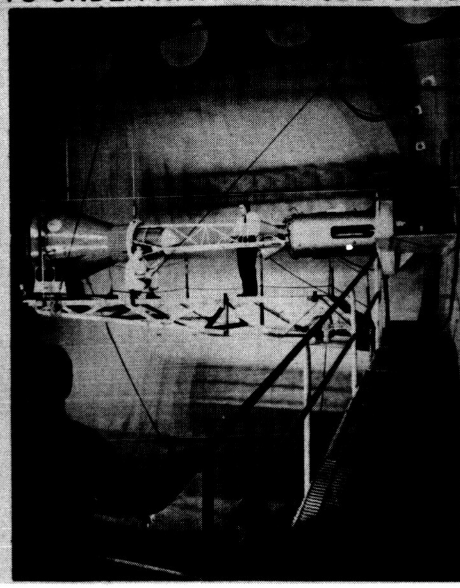


Figure 16 C-53287

ARRANGEMENT AT LANGLEY RESEARCH CENTER FOR VIBRATION EXCITATION OF A PRODUCTION MERCURY SPACECRAFT

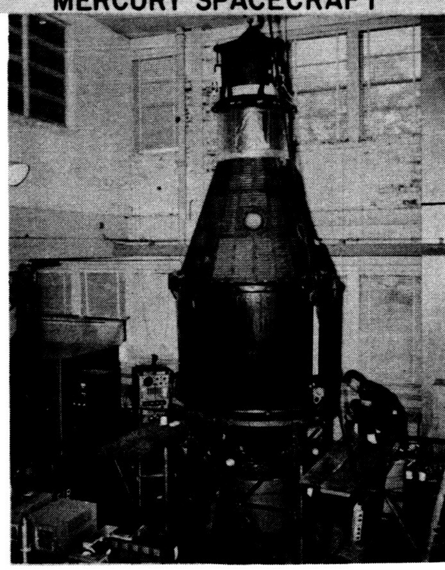
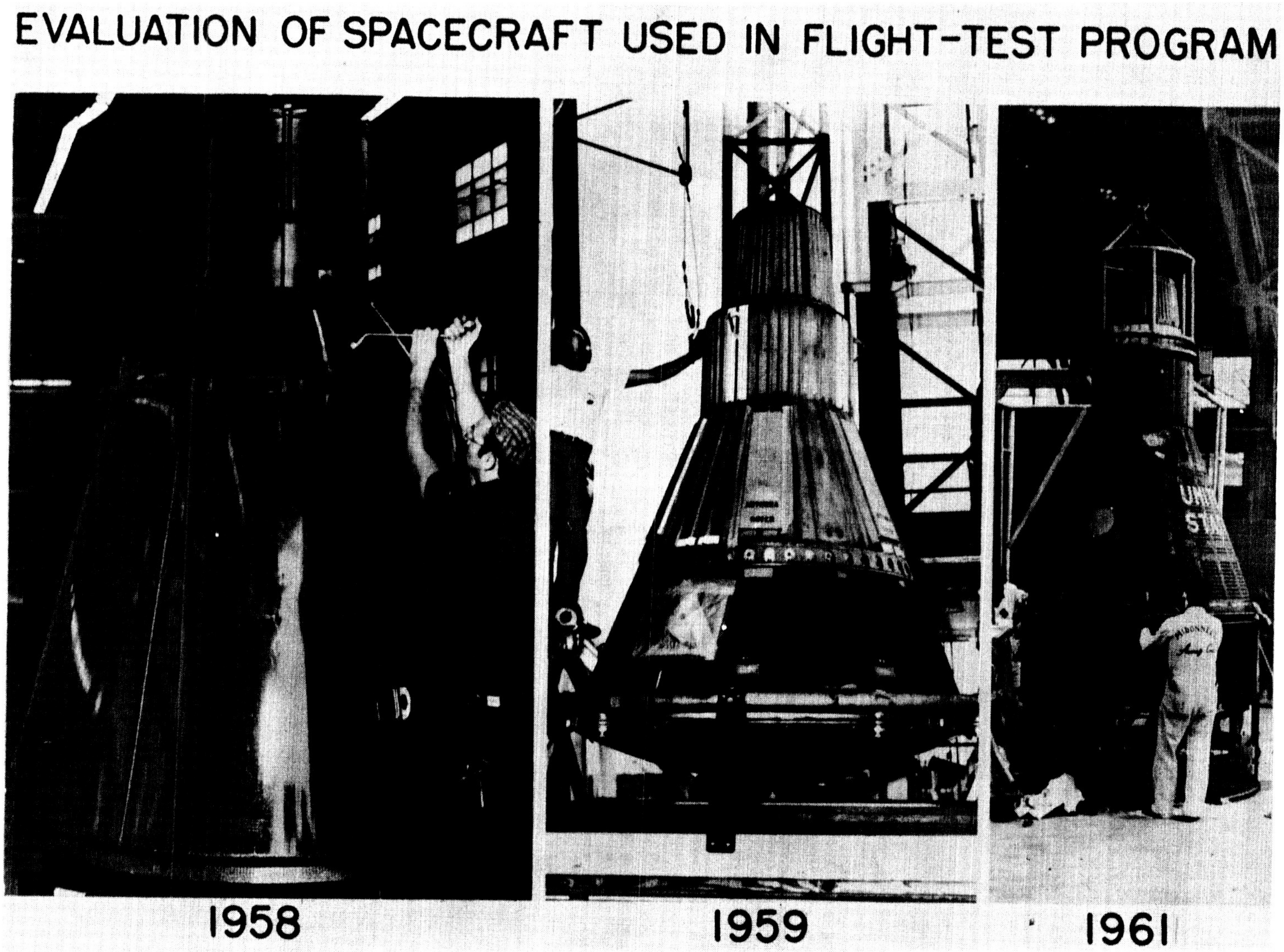


Figure 17 S-61-85



(a) (b) (c)

Figure 18

S-61-38

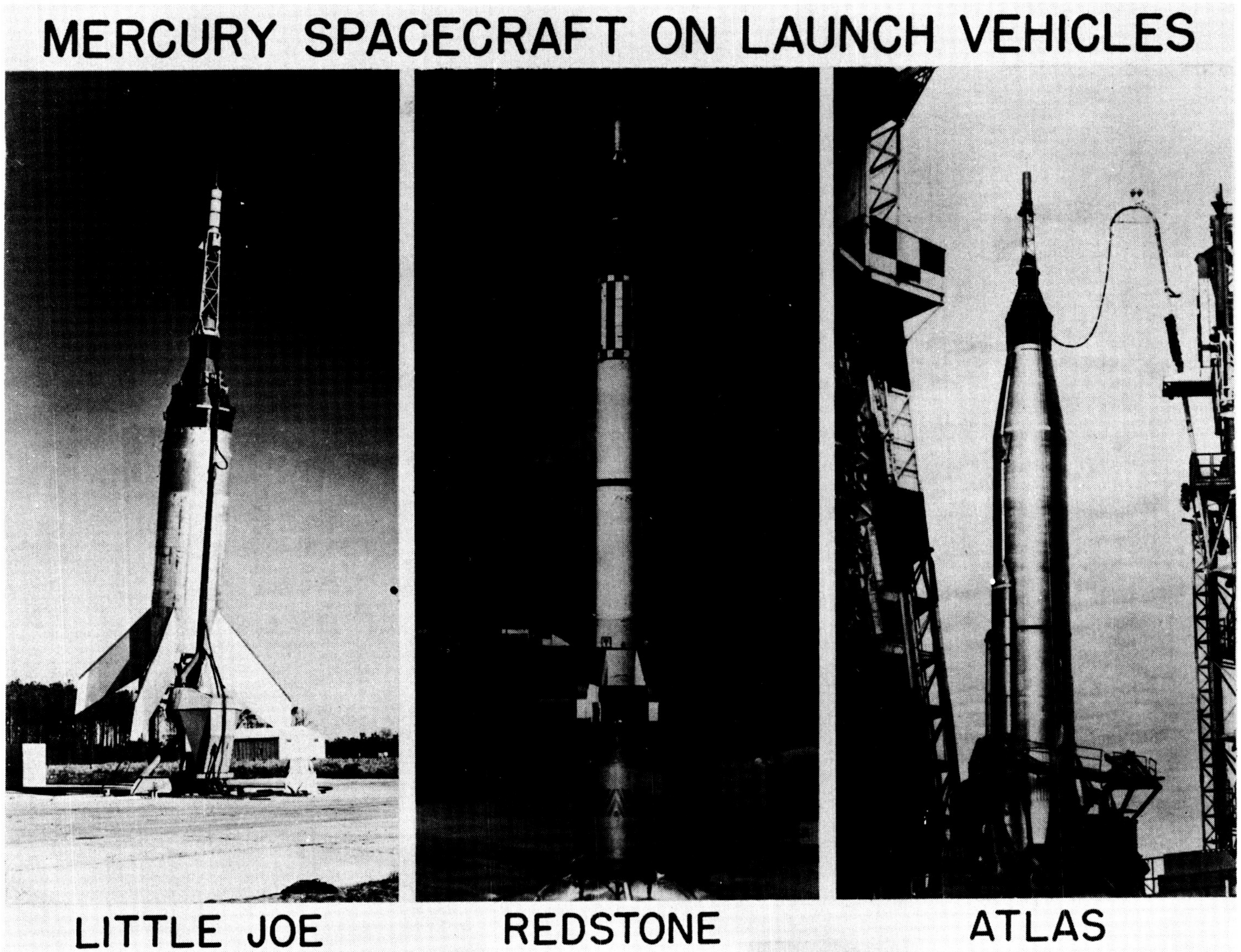


Figure 19

S-61-43

[redacted]

MERCURY SPACECRAFT—LAUNCH-VEHICLE STRUCTURAL COMPATIBILITY

By Robert E. Vale and George A. Watts

NASA Space Task Group

The intent of this paper is to recount those areas that were of concern in assuring the structural compatibility of the Mercury spacecraft with the Atlas launch vehicle. The Mercury-Atlas vehicle involves the use of an existing missile system as the launch vehicle for a manned satellite with minimum modification. The launch vehicle used is essentially a standard D-series Atlas. (See fig. 1.) The major structural components of the Atlas are the jettison or booster section and the tank section. The structure of the jettison section is comprised essentially of a cylindrical shell, in which the booster-stage propulsion is supported. This section is jettisoned at staging. The tank section is a 61-foot cone-cylinder consisting of a thin pressure-stabilized stainless-steel shell. The cone is closed at the forward end by a domed bulkhead. (See fig. 2.) At the junction of the bulkhead and the conical side skins, an attachment ring is provided for installation of the payload. The Atlas tank section being a thin-walled shell deriving its strength from internal pressurization is a highly efficient structure. In order that it will operate efficiently, however, particular care must be exercised to insure that all loads introduced into the tank section at the interface are ideally distributed. The adapter must be designed to meet this condition. The Mercury-Atlas adapter is a cylindrical semimonocoque structure. It is comprised of a thin continuous inner skin upon which is welded a corrugated outer skin. The corrugations are closely pitched to insure that loads introduced into the launch vehicle are evenly distributed. The corrugations are in turn stiffened by internal rings. The Mercury spacecraft and escape tower are described in a previous paper by Aleck C. Bond. The present paper deals primarily with the interface area. 38

In order to evaluate the structural compatibility of the spacecraft—launch-vehicle combination, one must consider loads, environments, and detail design of the interface. In each of these areas, one must be on the alert for any detrimental effects and interactions. The various sources of load including vehicle dynamics have to be reviewed and assessed for their structural significance. A major loading of the vehicle is caused by acceleration and drag, which produce large axial loads that occur on every flight and are accurately predictable. Venting, a source of load sometimes neglected, was found to contribute to the adapter and interface loading. This loading due to venting could have been large but was kept small by careful attention to vent hole area and location. The loads due to the roll maneuver are insignificant, since the vehicle has an extremely high torsional strength capability. However, the pitch program produces angles of attack with a consequent normal loading of the vehicle. The Mercury programmed pitch maneuver has been deliberately tailored to the mean winds at Cape Canaveral so that bending moments due to the maneuver and the mean winds will oppose each other.

The design loads due to winds and gusts are not consistently defined by all those engaged in space technology. It is, therefore, necessary to digress at this point in order to present the definitions of these terms as used in this paper. In the Space Task Group, it has become the practice to define gusts as perturbations that excite the oscillatory degrees of freedom of the vehicle. That is, during gust interception, the vehicle will experience rigid-body rotation and excitation of its elastic-body modes. Hence, gusts will be characterized by wavelengths equal to or less than those associated with the vehicle speed and its rigid-body frequency. On the other hand, wind is defined as a movement of air that produces nonoscillatory loads, which are quasi-steady in nature. The resultant motion of the vehicle will be a rigid-body translation and will not include rigid-body rotation. The profile of a wind so defined will be characterized by wavelengths larger than those associated with the vehicle speed and its rigid-body frequency. Balloon soundings only measure wavelengths of this magnitude. The quasi-steady loads acting on the vehicle are easily calculated from these balloon-measured profiles and the programmed pitch maneuver with no wind. With these definitions, it is possible to make a conservative allowance for gusts and then impose any required restrictions on the winds alone. In this way, it is feasible to use balloon-measured profiles near the time of launch to insure that restrictions are not exceeded.

Sufficient measured profile data pertaining to short wavelengths were not available for determining a satisfactory level of horizontal gust for use in vehicle design. Without these data, it was decided to determine the levels of gusts required to exceed a preassigned percentage of the vehicle strength and to see if these levels were much larger than those one would intuitively expect in nature. Both single and multiple gusts were investigated.

It was found that the spacecraft—launch-vehicle combination was not particularly sensitive to single large gusts such as those used in the design of airplanes (see fig. 3 where M_b is bending moment and T is time) because of the small lift effectiveness of the vehicle as compared with that of an airplane. The vehicle was, however, very sensitive to small repeated gusts at the structural natural frequencies because of the low levels of damping in the whole servoelastic system. Airplanes, of course, do not amplify the repeated small gusts because of the large aerodynamic damping provided by the wings. Analog studies have shown that a six-cycle sine gust could induce as much as eight times the bending moment induced by a single one-minus-cosine gust of the same amplitude.

In analyzing short-wavelength wind-profile excursions in the future, attention will have to be focused not only on the large singly occurring gusts of the type considered in the design of airplanes, but also on the small gusts which may be repeated many times. Since gusts alone can load the vehicle to 20 percent of its structural capability, a standard nationally accepted gust criterion is required. More horizontal gust data are needed to establish this criterion.

Buffeting is another source of load that is of concern. A question that was raised after the MA-1 failure was "Can the pressure fluctuations acting on the forward portion of the vehicle excite overall vehicle structural modes and cause large buffet bending moments?" By the use of conservative methods and wind-tunnel measured fluctuating pressures, large buffet bending moments were calculated. From the MA-1 instrumentation, it was not evident that large buffet bending moments had materialized. However, since these calculated buffet bending moments were large and the sum of all these loads was approaching the limit allowable loading of the vehicle, restrictions were imposed on the operating winds for the MA-2 flight. Again, the flight records failed to show any significant overall buffeting. Even though it is believed that the Mercury configuration breaks up the shed turbulence fine enough to prevent overall buffeting, future flight data will still be monitored for evidence of this type of load. When sufficient confidence is gained that this type of load is negligible, the imposed restrictions can be relaxed.

Experience has shown that loads due to transients at lift-off, staging, and tower jettison are moderate and since they occur when the vehicle is not under major load from other sources, they are not of concern. Loads due to hard-over engine gimbaling due to control failure are critical at the time of maximum acceleration just prior to staging when aerodynamic loads are negligible but structural temperatures are a maximum. These loads are predictable and can be catered for. Fortunately, such loads have not been experienced in flight.

In addition to the loads, the environments to which the vehicle is subjected must be considered. The two important environments for the launch vehicle are aerodynamic heating and noise. The aerodynamic heating, particularly of the adapter, has not been a problem in the Mercury program, because it has been considered from the beginning.

The temperatures and temperature gradients encountered during flight are moderate and their effects were minimized by the proper analyses, detail design, and selection of materials. By no means, however, can aerodynamic heating be neglected.

The noise environment was found to be the more serious of the two. Measurements of noise were made in some Little Joe flights earlier in the program. In one flight internal sound pressure levels were measured and external levels were estimated from these measurements. In another flight external sound pressure levels were measured directly. From these data the external noise was established as a maximum of 150 decibels. This value proved to be somewhat unconservative. It became apparent that the sound pressure level is not constant over the spacecraft-adapter region of the vehicle. The internal measurement could only be used to give a mean value, and the external measurement was misleading because it was not made at the point of maximum noise. Later in the program, fluctuating pressures were measured on a 1/3-scale model in a transonic wind tunnel, and sound pressure levels of 165 decibels and 160 decibels were measured on the spacecraft and on the adapter and conical portion of the launch vehicle, respectively. Figure 4 presents the sound pressure level at the adapter as a function of flight time derived from the tunnel test data. In this figure sound pressure levels measured at the adapter during the MA-2 flight are also shown. It may be seen that the two curves are in reasonable agreement. If the noise in the tunnel tests had been measured over a larger frequency range, it is expected that better agreement between these two curves would have resulted. The curves show that noise levels can be measured on models with reasonable accuracy. For purposes of comparison, the noise level that might be expected on a clean aerodynamic shape (0.006q, where q is dynamic pressure) is also shown. From this figure, one can see that the Mercury configuration may be termed very "noisy."

The effect of noise on the structural capability was found to be very important after the failure of the MA-1 vehicle near the time of maximum dynamic pressure. Many possible causes of failure were investigated and eliminated. Telemetered data persistently indicated that the failure was initiated in the interface area. Careful investigation of the loads and strength revealed nothing particularly severe existed. No overall vehicle dynamics were present and the angle of attack was near zero. Therefore, the loads on the vehicle were moderate and were only those that were easily predictable. Furthermore, the tower, spacecraft, adapter, and conical portion of the launch vehicle had been successfully static tested to loads which were much higher than those deduced for the MA-1 flight. Attention was then focused on the local structural dynamics. A study of the fluctuating pressures in the interface area was conducted and the results are shown in figure 4.

At the same time a vibration program was instituted for the spacecraft, adapter, and conical portion of the launch vehicle. Before this program only the spacecraft and tower had been vibration tested. During the vibration program, it was found that two adapter modes could be excited relatively easily. (See fig. 5.) These modes were the third and fourth ring bending modes combined in each case with the first corrugation bending mode. From the impedances measured in this test program and with the use of the sound pressure levels measured in tunnel tests, it was estimated that the amplitude response of the MA-1 adapter could be as large as one-half inch. This estimation assumes that perfect spatial correlation between the pressure field and mode exists.

This magnitude of response can seriously impair the structural capability in three ways. Large bending stresses are induced in the adapter rings. Local crippling of the ring flanges can occur with a resultant reduction of column stiffening provided to the corrugations. A more serious effect is the periodic reduction of the longitudinal stiffness of the bowed corrugations. Because of this reduction of longitudinal stiffness the longitudinal load concentrates at nodal lines in the adapter, thereby introducing concentrated loads to the launch-vehicle tank section. (See fig. 6.) This load concentration drastically reduces the capability of the tank section to carry overall loads as well as aggravates the large existing discontinuity stresses in the shell just below the interface. In addition, torsional deflections of the interface rings occur, further aggravating the discontinuity stresses. Consequently, low-cycle fatigue was quite possibly a cause of the failure. In order to eliminate these effects, it was decided to stiffen the adapter rings and increase the skin gages in the upper tank section. The MA-2 vehicle was flown with the stiffened rings but with a doubler installed instead of the increased skin gages. This flight was successful. Special instrumentation carried on this flight showed that the environment was capable of exciting the adapter, and the increased ring stiffness was adequate in suppressing this response.

In summary, the experience gained during the Mercury program has shown that:

(1) Venting must be considered.

(2) Horizontal gust data are required.

(3) Multiple gusts must be considered.

(4) Aerodynamic shape should minimize noise.

(5) Structures must not be sensitive to noise.

(6) Good detail design of interfaces is essential.

(7) Entire vehicle including control system must be considered as an integral unit.

(8) Loads instrumentation is advisable on early flights.

MERCURY-ATLAS

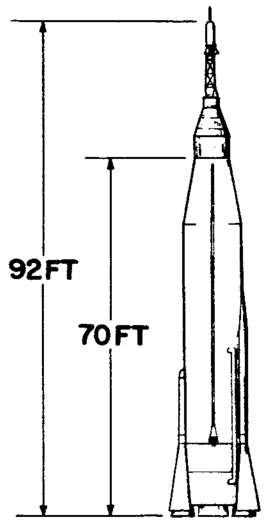


Figure 1

MERCURY-ATLAS INTERFACE AREA

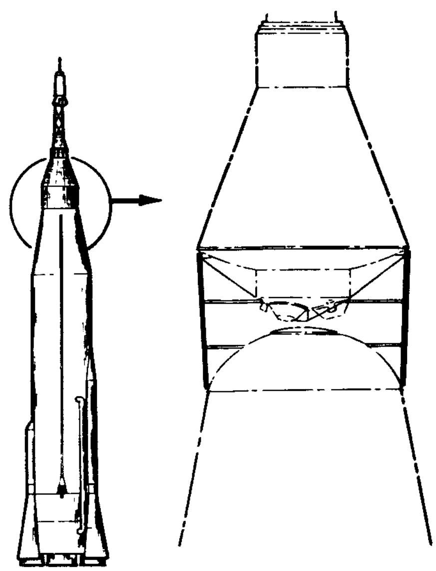


Figure 2

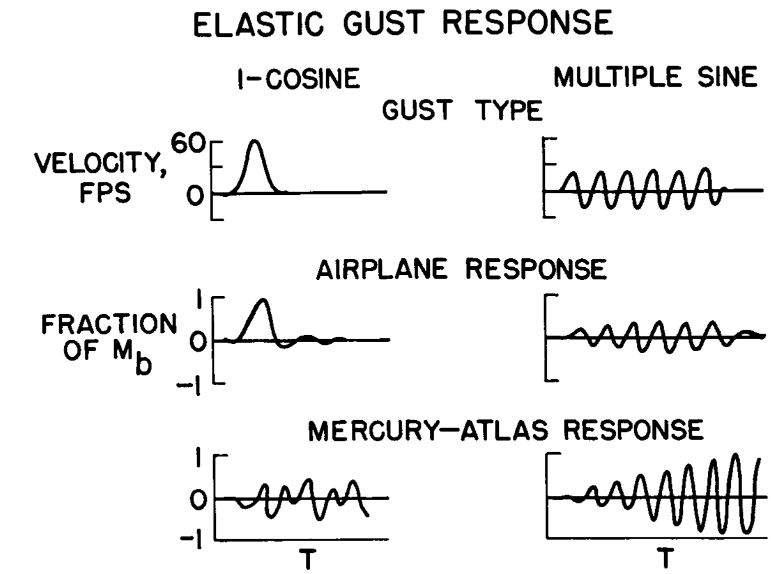


Figure 3

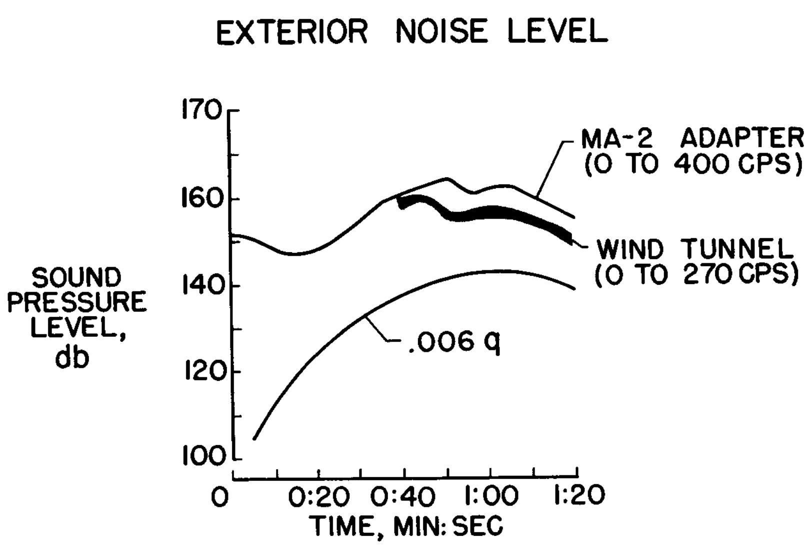


Figure 4

[redacted]

ADAPTER MODE SHAPES

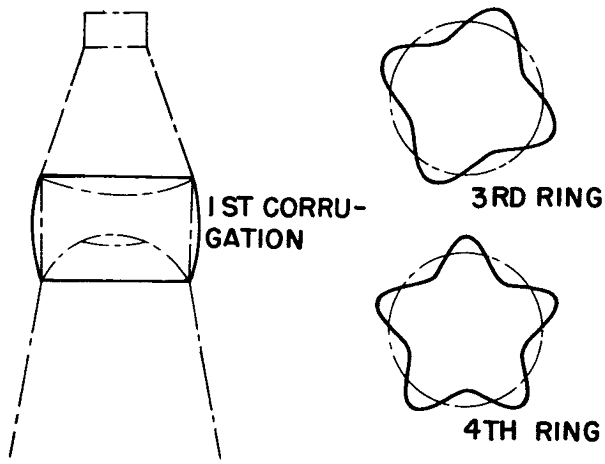


Figure 5

INTERFACE DETAIL

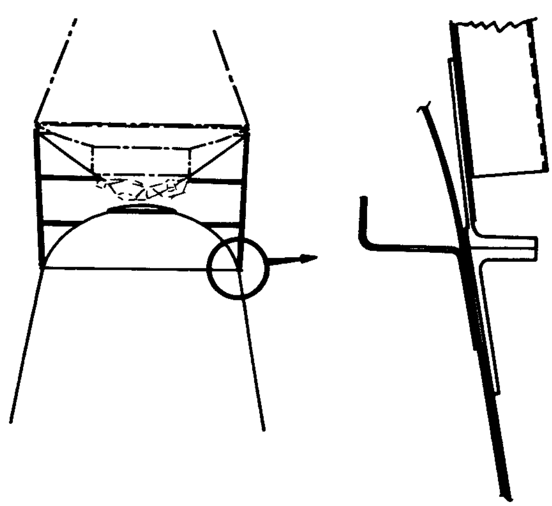


Figure 6

[redacted]

MERCURY SPACECRAFT—LAUNCH-VEHICLE

ELECTRICAL COMPATIBILITY

By Tecwyn Roberts, Paul C. Donnelly,
and Arthur Jonas

NASA Space Task Group

INTRODUCTION

The design philosophy and some of the problem areas encountered on the spacecraft—launch-vehicle interface of Project Mercury are discussed in this paper. The electrical interface involves the automatic abort sensing system; therefore, a description of this system is essential in any discussion on spacecraft—launch-vehicle compatibility.

AUTOMATIC ABORT SENSING SYSTEM

Project Mercury utilizes both Redstone and Atlas launch vehicles, which necessitated designing an abort system for each launch vehicle. The two systems differ in detail design because of differences in launch-vehicle characteristics, and hence the quantities monitored. However, the basic philosophy which governed the design of both systems is the same.

A number of factors were considered in determining whether the system should be fully automatic, or one that would display certain launch-vehicle parameters to the astronaut for evaluation and then have him initiate the escape action. An automatic system was decided upon. Analysis of prior Redstone and Atlas flight-test data indicated that the time interval between sensing a malfunction and catastrophic failure could be as short as 2 seconds; in this short interval it would be impossible for a human to evaluate and take action. There were also problems connected with the lack of space within the spacecraft for locating suitable displays, the increased weight resulting from the addition of a display, and the complexity of the launch vehicle and spacecraft interface connections added by the increased number of signals that would have to be provided. With an automatic system the abort package could be contained within the launch vehicle, with only control signals brought through the interface. Finally, the system had to provide escape capability for both an unoccupied spacecraft and the animal missions.

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The design requirements of the automatic abort system can now be specified, and are as follows:

1. Sense only catastrophic failures

2. Initiate an abort automatically when necessary

3. Avoid unnecessary aborts

4. Initiate an abort by the removal of a signal through the interface

5. Incorporate physical or effective redundancy in all sensor circuits

These design requirements were satisfied by careful selection of a minimum number of critical parameters to be sensed by the automatic abort system, from an extensive study of possible launch-vehicle systems malfunctions and from analysis of flight-test data. To guard against unnecessary aborts and still provide a high degree of reliability, the redundancy included in the system was governed by the most likely failure mode of the individual components. Finally the abort level of each parameter was carefully selected from flight-test evaluation and theoretical analyses. As additional test data become available, these levels are continuously reviewed in order to insure that current launch-vehicle development is reflected in the abort system. Table 1 shows the parameters selected and the abort levels for both the Atlas and Redstone launch vehicles. The parameters fall into two categories: explosive-type failures and control-type failures which would cause booster gyrations terminating in structural failure. In addition to the parameters shown, the loss of the 28-volt missile primary power supply would initiate an abort for both the Atlas and Redstone launch vehicles.

Considerable thought was given to the question of fire detection, both in the launch vehicle and in the spacecraft. Such a system was considered unnecessary for the launch because such systems were basically unreliable and also because several launch vehicles had been known to fly complete missions even with fires in the engine area. A fire warning system was not incorporated in the spacecraft because of the doubtful reliability of such systems and the fact that a fire warning system must be complemented by a fire extinguishing system, the use of which may result in more trouble than the fire. In Project Mercury the emphasis was therefore placed on making the spacecraft as fireproof as possible; in addition, the fact is that depressurizing during orbit is a pretty effective fire extinguishing system.

A more detailed description of the system and the parameters selected are given in references 1 and 2. However, a simplified block diagram of the system logic for the Atlas is shown in figure 1, where it can be seen that two pressure sensors in parallel are used for each parameter. Electrical continuity through the sensors into the abort-system network is required in the normal state. To initiate an abort both sensors of a given parameter must go to the open position. The sensors are utilized in pairs in this manner since it was established by test that the predominant failure mode was in the open position. It should be noted that, although this sensing circuit is dependent on a single power supply, the single power supply does not constitute a weak point in the system since this is the booster power supply and a power loss would necessitate an immediate abort. Two sets of rate gyros are used, the primary being the launch-vehicle gyros. The primary gyros set at the lower triggering level employ a long time constant (225 milliseconds) while the backup gyros employ a short time constant (80 milliseconds). The purpose of the difference in the two time constants employed with the two triggering levels is to guard against a low angular velocity over a prolonged period resulting in an excessive attitude buildup. Here the predominant failure mode was established to be in the no-output position. The normal operating position is no output, with a signal required to initiate an abort; therefore each sensor is independent. An abort level being sensed will result in the removal of a 28-volt signal through the interface, which causes the two launch-vehicle catastrophic-failure relays to drop out and hence initiate the escape sequence.

The system for the Redstone missile employs the same philosophy, but differs in the quantities measured, as shown previously, and the method in which the sensors are employed. An abort level being sensed is transmitted to an abort bus, which in turn causes the 28-volt signal through the interface to be removed.

In addition to launch-vehicle catastrophic failures there is one other aspect of the Mercury missions in which the abort system is utilized, namely, range safety considerations. The Range Safety Officer has the capability at any time during powered flight of commanding cutoff and missile destruct if necessary. A time delay of 3 seconds between cutoff and arming of the destruct package is incorporated in all Mercury launch vehicles. A Range Safety manual fuel cutoff would be sensed by the abort system, and would initiate the escape sequence. The 3-second time delay then provides adequate separation distance between the launch vehicle and the spacecraft if destruct has to be given.

In the initial flight-test phase when the system was flown in an open-loop configuration, some design deficiencies and human errors were encountered. In one instance the vented side of the pressure transducers was capped during inspection and erroneous readings resulted. In another case pressure-sensing lines had been strapped to lox lines, and freezing resulted. To date, experience during Mercury missions on both these systems has been very good. There have been two

Atlas and two Redstone closed-loop flights, and the system performed properly on all four flights. The first closed-loop Atlas flight was normal; the Abort Sensing and Implementation System (ASIS) monitored the performance and correctly did not initiate an abort. The second Atlas flight was destroyed by the Range Safety Officer at approximately 40 seconds after lift-off, and in this case the ASIS safely aborted the spacecraft after launch-vehicle cutoff, but before launch-vehicle destruct. The spacecraft was unharmed. The first closed-loop Redstone flight cut off early because of fuel depletion, and again the ASIS correctly initiated an abort. The second Redstone flight was normal and the ASIS correctly monitored the launch-vehicle systems without initiating an abort. These flights have exercised the ASIS in both its modes of operation, that is, in initiating an abort when required and in monitoring a normal flight without initiating an abort. This result clearly demonstrates the value of a careful study in selecting the abort parameters.

SPACECRAFT—LAUNCH-VEHICLE INTERFACE

The spacecraft—launch-vehicle interface is the focal point of coordination of all spacecraft and launch-vehicle systems. Guidelines under which the Mercury electrical interface was developed were that it would be simple, redundant, and would permit an early freezing of the engineering to allow the launch-vehicle design groups to concurrently design the mating circuitry.

An early question that had to be answered was, "How redundant would the circuitry be?" It was decided that the in-flight signals sent by spacecraft or launch vehicle would have a minimum of one path through each of two umbilicals. These umbilicals are connected to the spacecraft by explosive electrical disconnects and to the launch-vehicle—spacecraft adapter through mechanical disconnects. From the mechanical disconnects the launch-vehicle distribution was the responsibility of the launch-vehicle design groups. One launch-vehicle design group chose to terminate the redundancy immediately in the vicinity of the top of the launch vehicle because of the complexity of running additional wires to an already complex system. The other launch-vehicle design group did continue the redundancy to the receiving and transmitting function deep within the launch vehicle and made the circuits accessible.

Before the relative merits of the aforementioned approaches are discussed, the type of electrical interface signals between the Mercury spacecraft and its launch vehicle will be described. These signals are as follows:

1. Complex and launch-vehicle ground to tie in spacecraft ground

2. 28-volt signal from launch vehicle that holds in catastrophic-failure spacecraft relays which indicate no abort (abort bus not hot)

3. Catastrophic-fail detect relays which hold in engine cutoff relays (monitors ASIS ready or abort)

4. Spacecraft "Mayday" relay signal to cut off the launch-vehicle engine

5. Launch-vehicle engine separation signal

6. Time zero or lift-off signal

7. Sustainer engine cutoff signal

8. Scrub power signal (removes various systems from spacecraft electrical load)

9. Spacecraft-separation explosive-bolt firing signal

10. Squib-bus arm control

Experience to date shows that the method of running the redundancy far into the launch vehicle allows more utility in testing and isolating circuits. On one occasion it allowed a quick modification to one of two catastrophic-fail circuits at the main launch-vehicle distribution box and immediate test of the function after the change. Had this change been necessary in the other configuration, a delay of days instead of hours would have occurred.

One problem encountered in the early testing of the spacecraft—launch-vehicle interface was that in the abort mode the spacecraft explosive disconnects disengaged prior to the picking up of the engine cutoff relay of the launch vehicle. This problem was due to the fact that the response time of the explosive disconnects was so much shorter than the response time of a relay practical enough to complete the launch-vehicle function. Since in the normal mode there was no problem, the abort mode discrepancy was eliminated by backing up actuation of the engine cutoff relay by a spacecraft separation signal.

Another problem encountered was a mechanical discrepancy imposing a possible electrical failure. This problem again was experienced in the abort mode in dense atmosphere, when with the added acceleration of the escape rocket, the adapter fairings struck the explosive disconnects and did not allow them to separate from the spacecraft. The problem then became one of reentering the atmosphere with the interface umbilicals dangling. This problem occurred on the MA-3 flight. A photograph of the dangling umbilicals is shown in figure 2. The reentry heat could possibly fuse together a hot wire with a ground wire and blow fuses within the spacecraft and even de-arm the squib power circuit. The solution to the above problem was: (1) to redesign the fairings so that they would clear in the abort mode and (2) to remove voltages from the cables after spacecraft separation in case the cables did not disengage in any eventuality.

On the first Mercury-Redstone launch attempt, a launch-vehicle ground problem caused a malfunction that resulted in the launching of the escape tower by itself. At lift-off, the launch-vehicle ground wire was dropped prior to the signal lines, creating a potential between the launch-vehicle ground and the launch complex. The floating ground condition caused an engine cutoff signal to be transmitted to the booster engine cutoff relays, initiating the normal spacecraft separation sequence and thereafter energizing the landing and recovery systems. A system interlock prevented spacecraft separation. The spacecraft was undamaged. This occurrence clearly showed the necessity for providing a ground connection through each umbilical, or, as has now been provided, a lanyard-type disconnect for the launch-vehicle ground connection to insure that the ground wire is broken last.

The Mercury program has had a fairly clean bill of health as to radio-frequency (RF) compatibility of the spacecraft systems and the launch-vehicle systems, and only two problem areas have been encountered. The first was when a deviation from an assigned frequency occurred in one link of the spacecraft telemetry and RF incompatibility was encountered. The tests made prior to actually mating a spacecraft with a launch vehicle were done with assigned frequencies and did not impose any problems. However, because of delivery dates a telemetry transmitter was accepted in the first spacecraft with a lower transmitter frequency. The combination of this lower-frequency transmitter, the normal high-frequency transmitter, and the launch vehicle DOVAP system presented a beat frequency that blocked the spacecraft command receivers. Since this problem was only for one flight, the problem was solved by deleting the requirement for DOVAP on the first spacecraft launch. The second RF incompatibility experienced was in an early Mercury spacecraft Atlas launch-vehicle configuration when an interfering signal was generated by a beat of the two launch-vehicle telemetry links. This signal effectively blocked the ground telemetry stations from receiving the spacecraft low-telemetry link. This problem was readily solved by changing the frequency of one of the spacecraft telemetry transmitters. Subsequent spacecraft—launch-vehicle tests with specification equipment have not divulged any additional RF incompatibility.

CONCLUDING REMARKS

The relatively few problems encountered in the Mercury spacecraft—launch-vehicle interfaces result from their simplicity and the detailed testing that is accomplished at the launch site after mating the spacecraft to the launch vehicle. Each individual circuit is functionally tested, and the redundant paths are blocked during the testing. The fact that only one RF incompatibility has been encountered is due to the original hard look at frequency assignments and to the RF compatibility tests run at the launch site in full flight configuration.

The Apollo vehicle and launch-vehicle interface may well be quite complex because of the requirement for the crew to monitor launch-vehicle performance and possibly control launch-vehicle functions. The responsibility of the interface should be assigned at an early stage in development to one agent.

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TABLE I.- ABORT PARAMETERS

	Limit
Atlas launch vehicle:	
Lox tank pressure, psia	
Prestaging	21.5
Poststaging	11.0
Fuel manifold pressure, psia	
Booster engines	470
Sustainer engines	560
Sustainer hydraulic pressure, psia	2,000
Lox fuel-tank differential pressure, psid	2.5
Pitch and yaw rate, deg/sec	
Primary gyros	3
Backup gyros	4.75
Roll rate, deg/sec	
Primary gyros	6.4
Backup gyros	9.4
A-C voltage, volts	90
Redstone launch vehicle:	
Combustion-chamber pressure, psia	225
Pitch and yaw rate, deg/sec	5
Pitch and yaw attitude, deg	5.5
Roll attitude, deg	10.0
D-C control voltage, volts	50

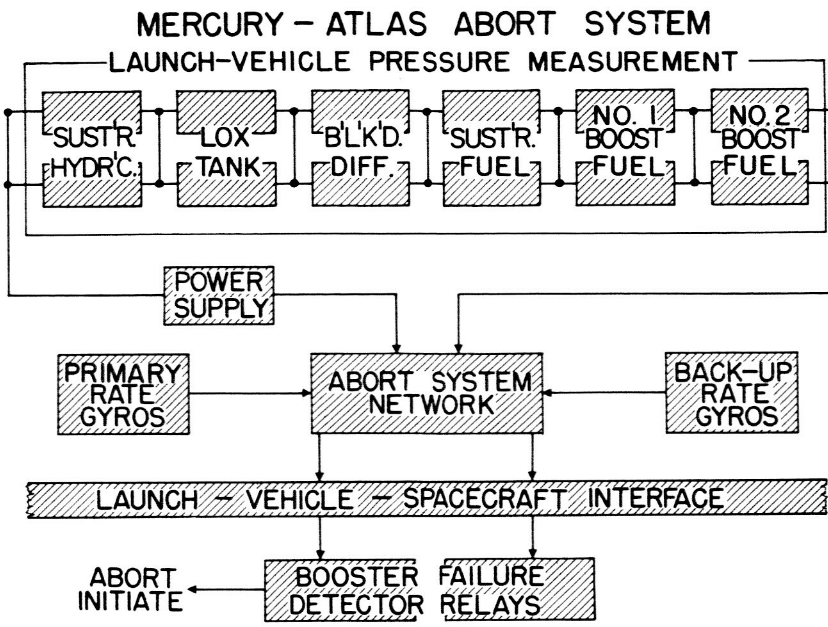

Figure 1

RESULT OF FAILURE OF UMBILICAL-CABLE RELEASE

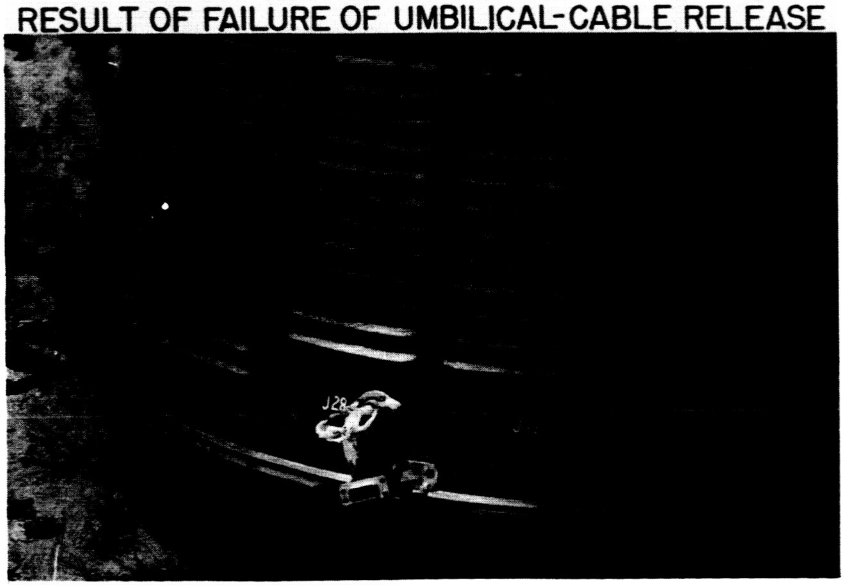

Figure 2

S-61-51

MERCURY LIFE SUPPORT EXPERIENCE

By Richard S. Johnston and Gerard J. Pesman

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INTRODUCTION

The Apollo spacecraft should be designed so that it reflects Project Mercury experience.

This paper summarizes the experience gained from development of the Mercury life support equipment with special emphasis on the test programs and unique problems encountered. The acceleration aspects of the Mercury flights are presented and the methods of confirmation or extension of these limits are described. The medical research aspects of the flights are reviewed.

LIFE SUPPORT SYSTEMS

The prime life support system is the environmental control system. This system provides a livable atmosphere for the astronaut. The system design requirements are summarized in table I. In summary, the system is required to have a minimum 28-hour space flight capability. The system supply was established by the flight duration and breathing oxygen requirements of 500 cc/min with a maximum cabin leakage of 300 cc/min. Carbon dioxide and manned heat outputs were established as 400 cc/min and 500 Btu/hr, respectively. The cabin pressure level was set at 5 psia with a pure oxygen atmosphere. Pressure suit ventilation flow was set at 10 cu ft/min at a pressure of 5 psia. Suit pressurization levels were set between 5.5 and 4.0 psia.

It was determined that a closed type of environmental control system best met these requirements. The Mercury environmental control system has been described in previous papers and therefore only a brief description is given. (See refs. 1 and 2.) Basically, the system is shown in figure 1. A pressure suit is provided for backup to the cabin pressurization system. A closed pressure-suit loop supplies breathing oxygen, provides body ventilation, and removes carbon dioxide, odors, water vapor, and heat. The cabin system maintains the pressure level between 5.1 and 5.5 psia with an oxygen atmosphere while in flight and controls the temperature within the cabin. An evaporative water heat exchanger is utilized for temperature control and oxygen is stored as a gas at 7,500 psi. System instrumentation provides the astronaut and ground flight controllers data on oxygen supply volumes and flow; cabin temperature and pressure; suit temperature, pressure and oxygen partial pressure. The environmental control system is designed to operate automatically with manual backup and controls for the more important system functions. Several normal and emergency modes of operation are provided.

The environmental control system, like other spacecraft hardware, was subjected to an extensive testing program. Figure 2 shows a breakdown on this test program. The system went through development, qualification, and reliability tests at the manufacturer's facility. Flight hardware underwent acceptance, spacecraft systems tests, and altitude chamber tests at Cape Canaveral, Fla., prior to each flight. In addition to these tests, a special manned test program was conducted with the environmental control system. A series of 12 manned tests was made with a production system in a boilerplate spacecraft, and the results were presented in a paper by J. A. Maloney and F. G. Richardson of the McDonnell Aircraft Corporation. In these tests, pressure and temperature simulation were provided. A variety of tests were conducted ranging from normal orbital flight of $4\frac{1}{2}$ and 28 hours to emergency modes of system operation. In addition to these manned tests, one test was made using a chimpanzee. These tests, although unique to the environmental control system, did provide valuable information on the spacecraft bioinstrumentation, pressure suit operation, and system operational procedures. Additional manned tests were run with the astronauts. A pressure suit circuit was installed in the U.S. Navy Johnsville Centrifuge and was used for both pressurized and nonpressurized Redstone flight simulations. Astronaut familiarization runs were made in the environmental control system boilerplate spacecraft providing the subjects with simulated flight experience on both normal and emergency conditions with combined pressure and temperature flights simulation. Prior to Astronaut Shepard's flight, better than 500 manned test hours were accumulated with the system. As a result of this manned test program, several significant system changes were made.

A few examples of these changes are included to illustrate the value of such a test program.

In the first series of manned tests, it was found that small leaks in the pressure suit circuit could result in the accumulation of nitrogen and a decay in oxygen partial pressure. This situation was possible since, at this stage in the program, a complete cabin oxygen purge was not made; and in certain discrete sections of the suit circuit, a slight negative suit-to-cabin differential pressure existed. To overcome this problem, several changes were made: First, a ground cabin purge to 100-percent oxygen was initiated; second, a small oxygen bleed was provided from the supply bottles into the suit circuit to insure a flushing action and to maintain a positive suit-to-cabin differential pressure; and third, the cabin oxygen partial pressure sensor was relocated into the suit circuit to provide continuous oxygen partial pressure data.

The test series also provided data on the adequacy of the snorkel postlanding ventilation system. In the Mercury system, postlanding ventilation is provided by the suit circuit compressor drawing ambient air through an inlet valve and circulating the air through the suit. The system design requirement was for a minimum 12-hour period in this mode of operation. An assessment of ambient conditions in the normal landing area and the heat input from the suit compressors meant that the astronaut could be ventilated with air at 97° F and a relative humidity of 60 percent. It was demonstrated in several manned and animal tests that these ventilation conditions were tolerable for the 12-hour period. From these manned tests and the flights to date it is known that the environmental control system can support manned orbital flights.

In summarizing the system development and tests, a few of the outstanding problems or experiences which would be of value to Apollo might be outlined:

(a) To date reliable methods of measuring carbon dioxide partial pressures by polarographic methods have not been achieved in system developments. Environmental incompatibilities, drifts in range, and poor response times were limiting usage factors. An unreliable reading in space flight monitoring is worse than no measurement at all.

(b) More precise physiological data on metabolic requirements is needed by engineering personnel for system design and evaluation.

(c) The number of openings through the pressure shell should be kept to a minimum. Use of a single opening for multipurposes should be the rule.

(d) System components requiring servicing should be accessible, easily removed, and require a minimum of checks after reinstallation.

(e) The ventilation system for the postlanding period, if required, should be designed to provide more satisfactory ventilation conditions. The temperature rise due to compressors or fans should be eliminated.

PRESSURE SUIT

The Mercury pressure suit was evolved from the U.S. Navy MK-IV full pressure suit (fig. 3). This suit is a single-piece garment which incorporates a ducted ventilation system. It is a single chamber suit with a vent inlet connection at the torso and an exit connection on the helmet. Communication equipment is provided in the helmet and a bio-connector is provided on the thigh. The suit is made up of two layers: a vinyl inner layer and a nylon outer cover. Suit leakage requirements were set at 250 cc/min at 5 psia. Several observations and comments on the suit development can be made:

(a) Thermal protection and ventilation were considered some of the most significant problems early in the suit development. The reentry heating pulse from orbital flight is shown in figure 4. From this curve it can be seen that the inner wall temperature rises to 270° F for a relatively short pulse and the cabin temperature rises to 180° F. To meet this requirement, several models of ventilation garments and the ducted system were integrated into the MK-IV suit and evaluated. It was found that the ducted system was adequate for this relatively short-term temperature rise. Integration of the inner ventilation garment with the exterior pressure shell was not possible because of excessive pressure drop in the ventilation system. The pressure drop in such an integrated suit varied from 10 to 15 inches of water as compared with 3 to 5 inches of water for the ducted suit.

(b) Suit stretch and loss in mobility were encountered after repeated pressurization to 5 psi. This problem was alleviated by adjustable sizing lacings and by undersizing the suits. It is evident that new and improved suit materials must be investigated in order to solve this problem completely. Recent suit developments indicate that this problem may be solved.

(c) Adequate pressurized mobility was achieved with the suit; however, improvement in wrist and hand mobility is still being pursued. In Mercury, only arm and hand mobility is required for capsule operation. In Apollo, suit mobility must be greatly improved.

(d) Suit comfort and habitability is adequate for the Mercury missions; however, longer duration flights dictate that new approaches to crew comfort, sanitation, and feeding must be established if pressure suits are to be utilized.

(e) The number of straps on the Mercury suit is considered excessive. Efforts should be made to minimize such straps to improve comfort, and to reduce interferences with various projections in the cabin.

The Mercury suits have been used extensively in the astronaut training program, and this information is currently being assembled and will be published in an NASA report.

ACCELERATION PROTECTION SYSTEM

The acceleration protection system consists of a support couch, restraint harness, and load attenuation devices. The support couch is shown in figure 5. These fiber-glass couches are contoured to fit each of the astronauts. The couch places the body in a supine position with respect to the flight loads and provides full torso and head support. Contoured panels for the thigh and legs are provided and are attached to the torso section of the couch. The couch is mounted in the spacecraft on a crushable aluminum honeycomb material designed to limit the couch loads to 40g under emergency landing conditions. The honeycomb is 6 inches in length and is distributed in columns under the torso section of the couch. Under emergency loads, the torso section crushes the honeycomb material. The leg support is hinged to allow the complete body to be supported as the torso section moves down.

The astronaut restraint system, figure 6, is designed to hold the astronaut in the couch. The majority of accelerations during flight and landing will force the astronaut into the couch. However, during certain aborts and possible tumbling after landing, the astronaut could be forced out of the couch. For these eventualities, a restraint system is provided. The system consists of a conventional lap belt and shoulder harness with the addition of a chest strap, crotch strap, knee strap, and toe cups. Head restraint has not been included in this system. The need for such a device was evaluated by track runs under abrupt decelerations up to 12g with the subjects wearing a Mercury full pressure suit.

The impact air bag system previously described provides additional attenuation for the landing phase. This device was incorporated into the spacecraft when it became evident that large horizontal velocities might impose injurious loads on the astronaut during ground landings. Design parameters on the bag were selected so that spacecraft accelerations would not exceed 20g longitudinal to the spacecraft or 10g lateral. These design limits are considered conservative and do not represent tolerance limits of man.

With this description of the acceleration protection system, some of the normal and emergency flight loads are reviewed. The significant loads encountered in the Redstone ballistic flights are shown in figure 7. At launch in the normal flight trajectory, the accelerations build up to 6g longitudinal to the spacecraft and then decay rapidly to weightlessness. After approximately 5 minutes, the spacecraft reenters the atmosphere and the loads build to 11g. The drogue and main parachute deployments impose small decelerations, and landing with the impact bag deployed will cause an abrupt deceleration of 10g to 15g dependent on the landing conditions. These flight load conditions are well within known tolerance limits as confirmed by the recent successful manned flight.

Several aborts can occur in the Redstone mission. The first of these is a pad abort. In this maneuver escape rockets are fired to remove the spacecraft from the launch vehicle and launch complex area. The thrust of the escape rockets produces a load of approximately 16g for 1 second. The probability of a ground landing following a pad abort is quite high due to the landward prevailing winds at Cape Canaveral. Ground landing loads with the impact bag deployed will produce loads longitudinal to the spacecraft of 10g to 15g and lateral loads under 5g depending upon winds and other landing conditions. If the impact bag fails to deploy for ground landings, the crushable honeycomb material will limit the loads to 40g. Lateral loads will not be attenuated and will be dependent upon the wind and other landing conditions. An animal drop program was successfully conducted at McDonnell Aircraft Corporation to evaluate the adequacy of the honeycomb material. This test program partially justified acceptance of these emergency landing loads. Recent human drops at Wright Air Development Division (WADD) have provided additional data that further justify these emergency landing conditions. In this program, subjects were dropped at 30 ft/sec by using crushable honeycomb material. To date, human subjects have successfully withstood loads greater than 40g at high onset rates. (See ref. 3.)

A second abort that imposes high g loadings can occur under escape conditions at high dynamic pressure. Under this abort condition, the escape rockets are used to separate the spacecraft from the launch vehicle. The rockets impart approximately 12g longitudinal to the spacecraft and at burnout the aerodynamic drag will cause a negative load of approximately 8g. During this abort maneuver, the astronaut will at one moment be forced into the couch and then an instant later will be suddenly flung out against the restraint harness. The same sequence occurs in either Redstone or Atlas missions. This abort condition has been successfully evaluated in a Little Joe flight using a rhesus monkey. The loads have been partially evaluated by the astronauts during an early centrifuge program. The subjects were subjected to positive 9g accelerations (eyeballs in) and the centrifuge gondola was turned and the 9g loads were then applied in the reverse (eyeballs out) direction. The simulation was not completely valid due to the time required to turn the gondola.

In the recent Redstone animal flight, the chimpanzee using a scaled-down version of the restraint system withstood loadings up to 18g due to a launch-vehicle malfunction which actuated the escape rockets and imposed the added thrust of these rockets. The animal continued to perform following this abort condition which indicated that the animal remained conscious throughout this high stress period. (See ref. 4.)

The reentry accelerations which reached 15g were above normal because of the added spacecraft velocity. The animal showed some decay in his performance rate following reentry.

The significant accelerations to be experienced in the orbital Atlas flights are shown in figure 8. Two acceleration pulses of 6.7g and 7.6g are experienced during launch. Following orbital flight the reentry loads peak at approximately 7.4g, and parachute deployment and landing loads are the same as shown for the Redstone flights.

The emergency aborts in the Atlas flights can be divided into pad aborts, high q aborts, and preinsertion and postinsertion aborts. The pad and high q abort loads are similar to those described in the Redstone flight discussion. Certain aborts can occur in prestaging or preinsertion that can cause loadings as high as 22g. Early in the program, this high g condition was successfully evaluated at Johnsville, Pa., using a contoured couch developed jointly by the U.S. Navy and NASA.

ACCELERATION PROTECTION SYSTEM TESTS

The most extensive of these tests using the Mercury system has been the centrifuge programs. To date, one Atlas and two Redstone programs have been conducted.

The last Redstone centrifuge program can be cited as an example to show the scope and results obtained. In this test series, astronaut training was the prime objective; however, other objectives were met. First, the flight biosensors and monitoring procedures were evaluated. Mercury range medical monitors were trained, by using a prototype range console. Second, the astronaut's personal flight equipment such as the pressure suit and restraint system was evaluated. Third, operational procedures and timing were established for astronaut flight preparation and ingress into the spacecraft. Fourth, astronaut physiological stress control and baseline information for interpreting in-flight and postflight data was obtained. Use of these data was illustrated by the recent MR-3 report. (See ref. 5.) During this program, only flight hardware, including the environmental control system, was utilized. Simulated Redstone flights were made with the suit pressurized and unpressurized at 5 psia. The astronaut check-out and transfer trailer was used for astronaut preparation and valuable information was accumulated on flight operational procedures. As a result of this program, certain equipment changes were made, astronaut reliance in the suit and in the environmental control system was gained, and the life support systems were accepted as flight-ready.

The results of these centrifuge programs have provided and will continue to provide experience and data which will be useful in the Apollo program. In the past, as a part of astronaut training programs, studies of the effects of reduced oxygen pressure on acceleration tolerance were conducted and showed no marked effects. Studies on spacecraft lighting to evaluate vision under acceleration were carried on. In the next program, it is planned that blood-pressure measurements in addition to electrocardiogram, and respiration and body temperature measurement will be made. If possible, metabolic measurements under acceleration will also be conducted.

In summarizing this discussion of the acceleration protection system, a few comments can be made that apply to the Apollo program. First, the design of the acceleration protection system in Mercury reflects the emergency acceleration conditions and not the normal loads. The design of the Apollo system must likewise be designed for these extreme conditions. Second, the Mercury restraint harness has proven satisfactory; however, the number of straps and adjustments make ingress and egress difficult. In the Apollo program, new and unique approaches to this harness problem should be developed. The use of an integrated restraint garment integral to the flight clothing or pressure suit may be the best approach. Third, the use of especially molded couches for each individual astronaut imposes handicaps on the operation and servicing at Cape Canaveral. Better than 4 hours are required to change a couch. In Apollo, use of a universal couch or liner is considered essential.

MEDICAL RESULTS OF MERCURY

In Project Mercury, a medical research program has been established. The prime physiological evaluation will be that of human exposure to periods of weightlessness for up to $4\frac{1}{2}$ hours. Programs of preflight and postflight physical and psychological examination are being employed to evaluate the flight stresses. In-flight recordings of the electrocardiogram, respiration rate, and body temperature are being taken for monitoring and postflight analysis. In later flights, it is planned that blood-pressure measurements will be taken. Studies of blood and urine chemistry are being made, evaluated, and correlated with control data to evaluate further the combined flight stresses. In-flight feeding under weightlessness is being evaluated through the use of paste and solid foods. The trace gases given off by the astronaut or system components while in flight are being determined through analysis of an activated charcoal filter. This listing of medical research programs is not complete or final. As flight duration and experience dictate, variations and additions to the programs are being made to provide as much biological data as possible.

The Mercury flight program will provide medical data and information upon which Apollo must be established. The research goals and objectives of the early Apollo flights will in part be dictated by these biomedical results.

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[redacted]

TABLE I

ENVIRONMENTAL CONTROL SYSTEM

	Requirement	System provision
Flight duration	28 hr	[a]31 to 35 hr
Oxygen supply	4 lb	8 lb
Metabolic O_2	500 cc/min	>10 liters/min
Cabin leak	300 cc/min	1,500 to 2,500 cc/min
Pressurization level	5 psia	5.5 to 4.0 psia
Oxygen partial pressure	5 psi	5.5 to 4.0 psi
Suit circuit heat production	1,000 Btu/hr	1,000 Btu/hr
Metabolic	500 Btu/hr	700 Btu/hr
Equipment	300 Btu/hr	300 Btu/hr
Suit ventilation flow at 5 psi	10 cu ft/min	11.5 cu ft/min
Carbon dioxide output	400 cc/min	>400 cc/min

[a]Additional coolant water required.

THE MERCURY ENVIRONMENTAL CONTROL SYSTEM

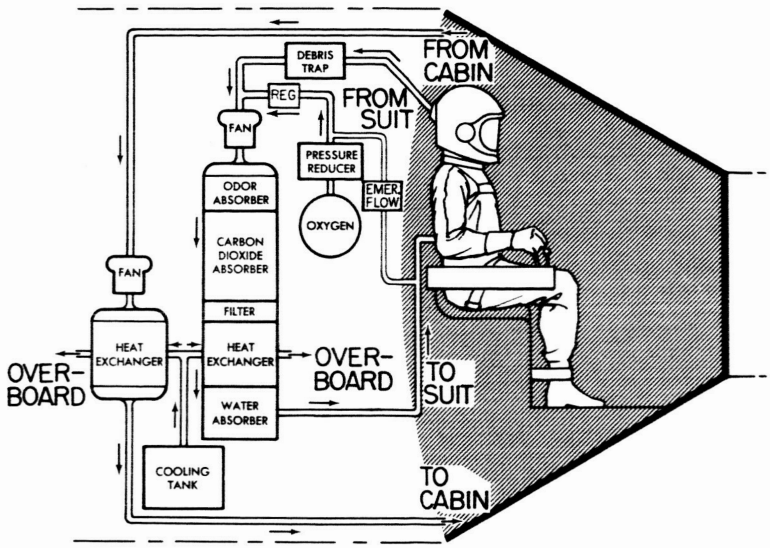

Figure 1

TEST PROGRAMS FOR ENVIRONMENTAL CONTROL SYSTEMS

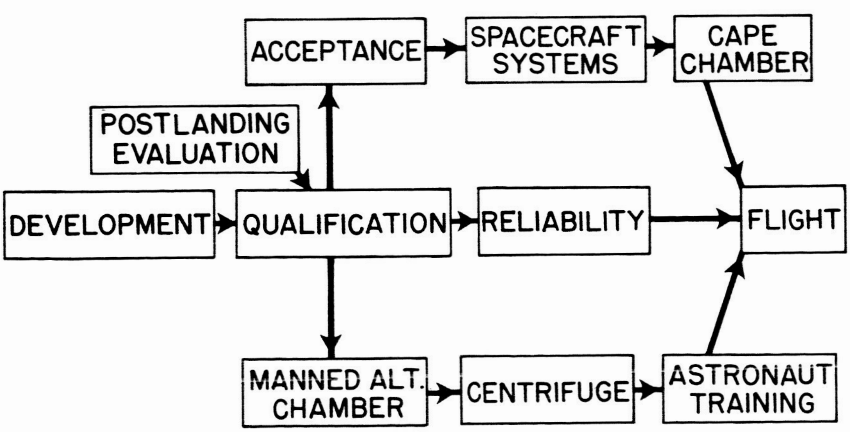

Figure 2

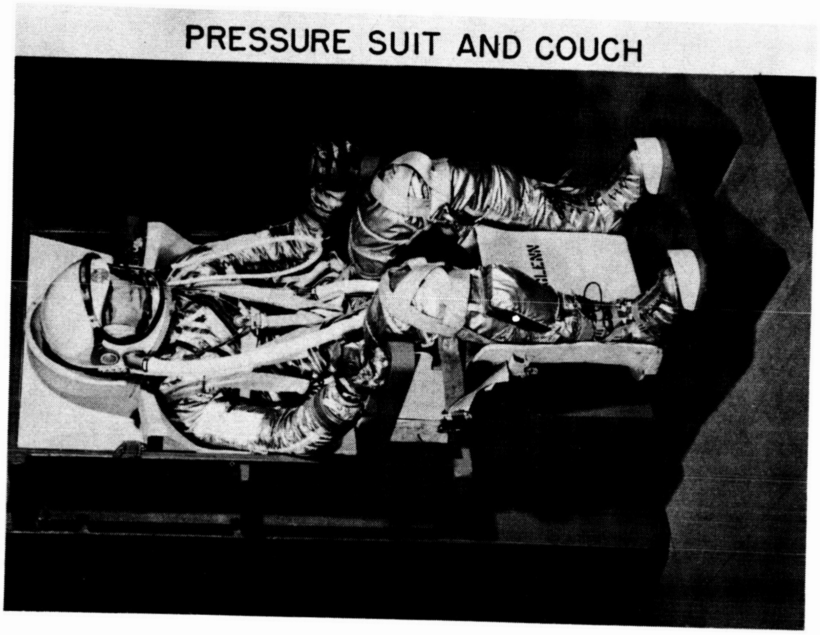

Figure 3 S-61-53

REENTRY AND POSTLANDING TEMPERATURES

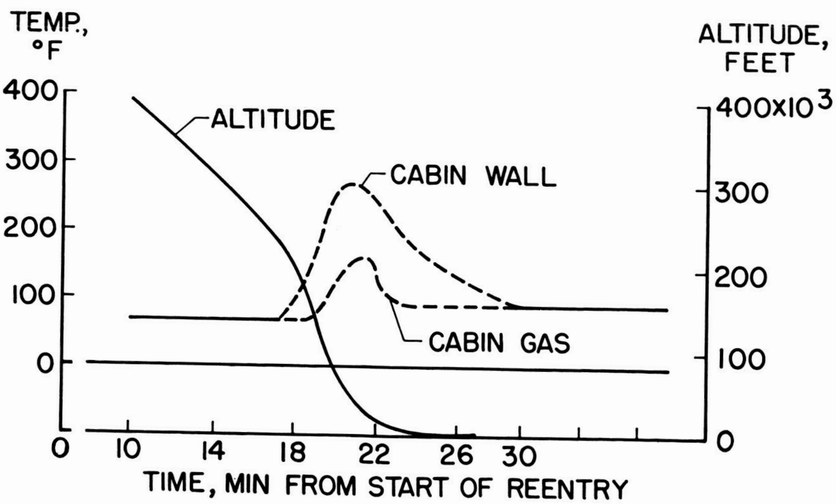

Figure 4

[redacted]

ASTRONAUT'S SUPPORT COUCH

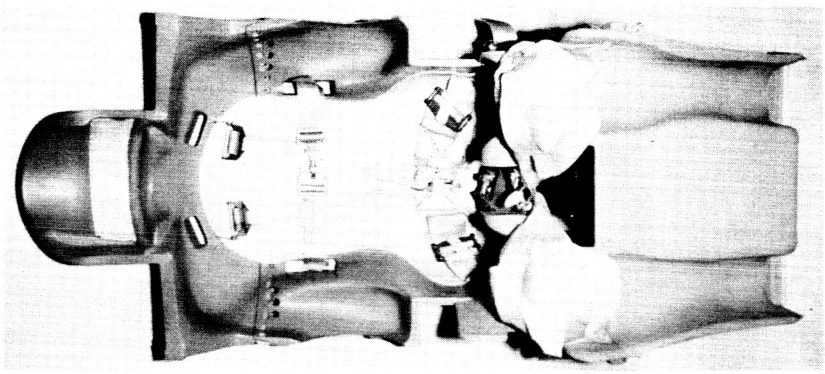

Figure 5 S-61-54

ASTRONAUT'S RESTRAINT SYSTEM

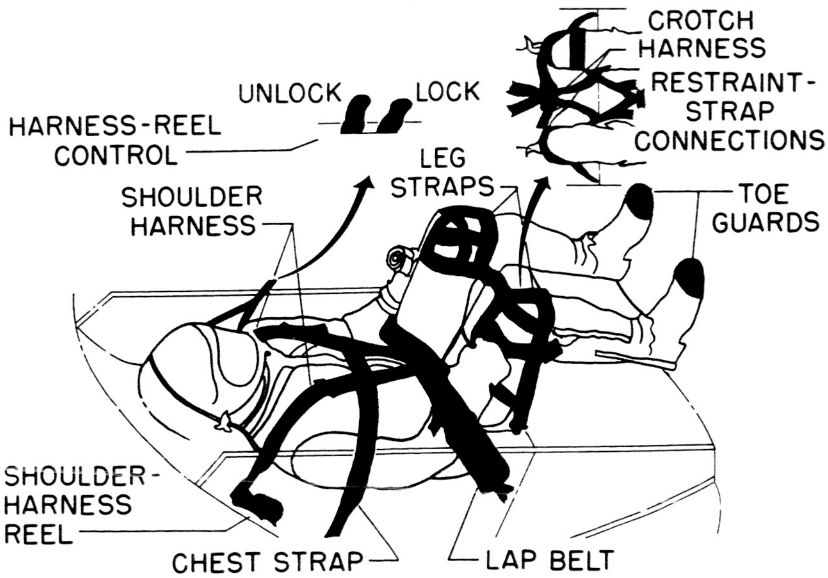

Figure 6

MERCURY-REDSTONE ACCELERATION TIME HISTORIES OF NORMAL MISSIONS AND q_{MAX} ABORTS

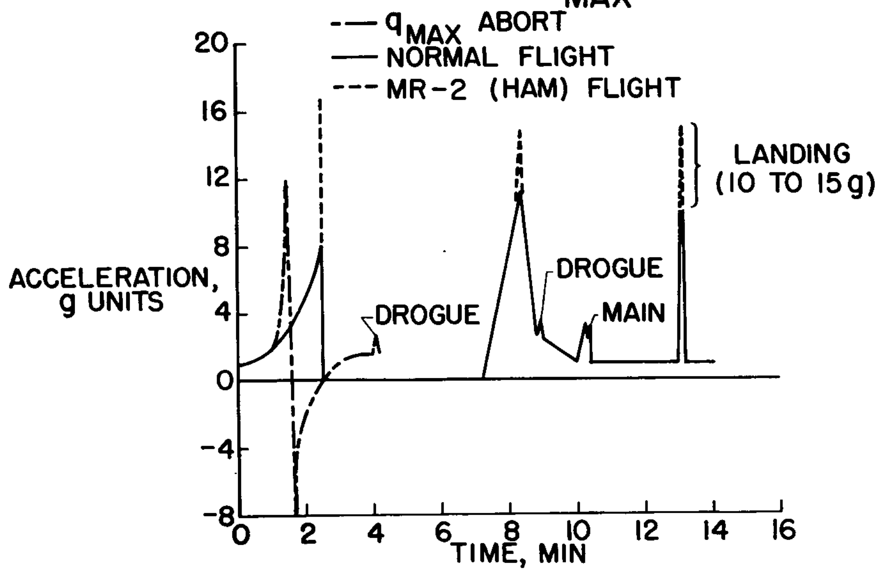

Figure 7

ATLAS NORMAL-ACCELERATION TIME HISTORY

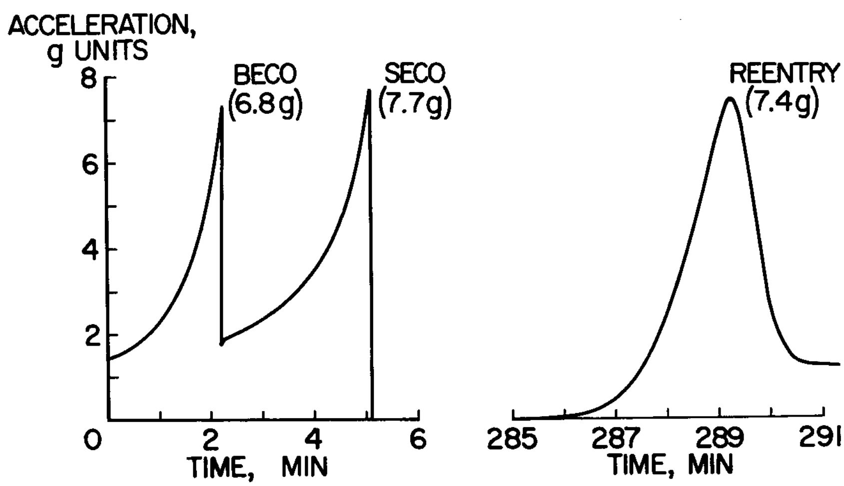

Figure 8

IX. PROJECT MERCURY OPERATION AND TRAINING

MERCURY OPERATIONAL ANALYSIS

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This paper will discuss mission analysis studies which have proved useful in Project Mercury. Although the same problems do not specifically apply to the Apollo program, there are similar problems for Apollo which will have to be solved.

In figure 1 are shown the elements which enter into operational mission analysis. The spacecraft, launch vehicle, ground system, and operational considerations must be considered as an entity. Such operational aspects as launch operations, flight control, abort considerations, environment, landing and recovery, and the human system must be given consideration in the design or analysis of a given mission. The spacecraft and the launch vehicle must, of course, be considered and include such elements as performance, guidance and control, and system limitations. Equally important, however, to a complete mission analysis is the relationship of the ground systems and operational factors. For example, it is only by combining all these elements that the final trajectories are chosen.

Some of the mission analysis studies conducted for Project Mercury are outlined as follows:

I. General studies
   (1) Studies of geophysical data
   (2) Selection of Mercury orbit
   (3) Insertion limitations (go—no-go)

II. Launch-vehicle studies
   (1) Development of explicit guidance equations
   (2) Shape launch trajectory
   (3) Abort sensing criteria

III. Spacecraft studies
   (1) Abort studies
      (a) In-flight control
      (b) Miss distances and lateral loads for escape-rocket aborts
      (c) Miss distances after poststaging aborts
   (2) Orbit studies
      (a) Dispersion in retrotime
      (b) Navigational aids for astronauts
      (c) Orbit lifetime

(3) Reentry and recovery studies
   (a) Abort landing-area control
   (b) Retroattitude and sequence optimization
   (c) Dispersion areas

IV. Computing and tracking studies
   (1) Real-time computing
      (a) Displays for in-flight monitoring and control
      (b) Orbit determination
      (c) Acquisition data to tracking sites
   (2) Tracking studies
      (a) Antenna look angles
      (b) Optimum spacecraft orbit attitudes
   (3) Prelaunch operational computing
      (a) Actual wind effects on launch-area abort landing points
      (b) Actual wind effects on launch-vehicle—spacecraft in-flight loads

Among the general studies conducted were those concerned with geophysical characteristics such as the determination of the best estimates of the atmosphere and geodetic constants. Use was made of the latest available satellite data in this study. Considerable effort was also made in the selection of the Mercury orbit. This included the selection of the ground track and the selection of the actual insertion conditions which determine apogee and perigee. The final insertion conditions chosen were based on launch-vehicle performance, guidance accuracy, spacecraft performance, and certain operational considerations. Another important study was the determination of go—no-go type of insertion limitations.

Among the launch-vehicle studies made were the development of the guidance equations, the shaping of the launch trajectory, and the determination of abort sensing criteria. The guidance equations developed were explicit in that they are independent of engine performance, launch-vehicle or spacecraft weight, the time of launch, and atmospheric effects. The launch trajectory was shaped so as to maximize performance and minimize abort loads. Abort sensing criteria had to be determined for automatic abort sensing based on missile measurements and trajectory deviations based on tracking measurements.

Abort studies make up a large percentage of the mission analysis studies. Studies were made to provide flight controllers with the knowledge of when to initiate aborts for maximum pilot safety. Studies were made to obtain safe miss distances between the launch vehicle and the spacecraft and to reduce lateral loads in escape-rocket aborts. The escape-rocket offset was selected on the basis of a compromise between high lateral loads and low miss distances. Also of importance were miss distance studies for aborts after staging where retrorockets are fired. Here the effects of sustainer-engine tail-off, posigrade, and retrograde effects had to be considered. Of course, normal mission miss distance studies were also included.

Included in studies of orbital flight were retrotime dispersion studies based on guidance and control errors, calculations for navigational aids for the astronauts, such as star charts, and orbit lifetime calculations.

Reentry and recovery studies included abort landing area control in which retrocontrol is used to minimize the number of recovery areas, the determination of the optimum retroattitude and sequence for firing, dispersion landing area calculations, and studies to obtain optimum locations for recovery forces.

Real-time computing has proved to be very valuable to Project Mercury for use in providing displays for in-flight monitoring and control. The exact orbit must be determined from tracking data from the worldwide network, and information for tracking sites must also be computed. Tracking studies included the determination of quality of tracking which the stations would receive and recommendations concerning optimum spacecraft attitudes.

Other computing tasks which have proved valuable are the calculations of the effects of the actual wind profile measured immediately before each flight to determine wind effects on launch abort landing areas and loads.

Up to this point, the general analysis studies conducted for Project Mercury have been described. As flight dates approach, it is often necessary to update many of these studies by incorporating the latest weight and performance information.

The computations necessary for flight readiness are (1) the computation of the nominal and abort trajectories based on the latest weights and performance, (2) calculations of specific data concerning pilot safety for use by flight control personnel, (3) range-safety trajectories, (4) information for the tracking sites, (5) sighting data for recovery forces, and (6) latest landing and dispersion data for recovery forces.

Next some specific Project Mercury mission analysis studies will be discussed. In figure 2 is shown the manner in which Atlas performance, guidance accuracy, and operational considerations affected the selection of the orbital insertion conditions. In this figure the time of booster-engine cut-off is plotted against the insertion altitude. The performance is given in terms of an excess velocity (ΔV).

This velocity is defined as the velocity that would be obtained if the engine was not cut off but allowed to burn to fuel depletion minus the velocity which is desired for orbit ($\Delta V = V_{\text{fuel depletion}} - V_{\text{planned}}$). The performance line in this figure represents an acceptable minimum performance for Project Mercury; therefore, anywhere below the line would result in an acceptable performance.

The next factor to consider in the selection of the orbit is guidance accuracy. Since the Mercury-Atlas is guided by ground-based radio guidance, insertion into orbit must be such that the line of sight from the guidance site to the vehicle is always above the horizon. The accuracy is degraded significantly when the elevation angle is reduced below 8° to 10°. The line for the minimum acceptable elevation angle is also shown in figure 2.

Next, however, operational requirements must be considered. The operational considerations for Mercury are as follows: (1) Have the ability to avoid Africa for near-insertion aborts and (2) have an acceptable lifetime. Lifetime has been accounted for in this figure in that the excess velocities are given for a constant lifetime.

The first requirement is that an abort from the go—no-go velocity (or the minimum acceptable velocity) would impact short of Africa. This is the line (shown in fig. 2) of the minimum acceptable landing distance from Africa if an abort was initiated at the minimum acceptable velocity for orbit (go—no-go). The region of acceptable orbits is shown by the triangular area in the figure. The conditions selected are then chosen within this triangle to obtain maximum performance.

As indicated, one abort recovery area (E) is located a safe distance off the coast of Africa, and figure 3 indicates the manner in which other abort recovery areas were selected. In order to minimize the size of the recovery forces it was desirable to restrict the recovery areas to discrete areas by using the retrorockets for control. Shown in figure 3 is the landing longitude for aborts initiated at various velocities. The bottom line represents the landing points for aborts in which the retrorockets are fired as soon as the spacecraft completes its turnaround maneuver after insertion. The top line represents the landing points where the retrorockets are fired after a maximum time delay. The maximum time delay for retrofire is based on firing the retrorockets before significant heating occurs. In addition to these two limitations (minimum time delay and maximum time delay), there is also another operational consideration. This is the requirement that all retrocommands should be given within an acceptable range from Bermuda. The line which represents an acceptable command range from Bermuda is also shown in figure 3.

[redacted]

As was indicated previously, area E is selected on the basis of landing a safe distance off the African coast. It can be seen that the spacecraft could land in area E over a range of speeds (as shown in fig. 3) by varying the retrotime between the minimum and maximum acceptable retrotimes. In a similar fashion, other recovery areas (D, C, B, and A) are selected. Below a velocity of about 23,000 feet per second, the retrorockets become less effective as a range control and a continuous recovery effort becomes necessary. Also shown in this figure is a scale of time to go until engine cut-off. It may be noted that these recovery areas are transversed in a very short time, a matter of 2 seconds or less for each recovery area. The retrotimes vary from a time of about 30 seconds up to several minutes in length, so it is obvious that to determine the retrotime in the case of an abort, real-time computing is necessary and is based on the most accurate sources of tracking. This real-time computing capability is available for the computation of retrotimes as a function of the position and velocity at cut-off.

It appears that the abort recovery problems on Apollo will be considerably more complex than for Mercury and much attention must also be given to operational problems other than the effects of heating and loads.

Next, the criteria and methods used to determine the acceptability of the Mercury orbit will be discussed. In figure 4 is shown the operational go—no-go criteria for acceptable orbits. The minimum orbit is an orbit in which one orbit could be completed safely from the standpoint of satisfactory heating, loads, and recovery area considerations. Shown in this figure is a plot of the flight-path angle (of the velocity vector above the horizon) against the inertial velocity. Since the atmospheric density and the drag coefficient are really not known very accurately, the acceptable orbit line, instead of being a line, is actually a broad area and, therefore, the selection of the minimum acceptable velocity is approached on a probability basis. An estimate has been made of the conditions where there would be at least a 99-percent certainty that one orbit could be completed safely. These conditions are indicated by a line in figure 4. The variations in the atmospheric density and drag coefficient were made with the use of available satellite data.

The maximum-energy acceptable orbit is based on obtaining a safe reentry from all points in the orbit, taking into consideration recovery areas, heating, and loads. As the velocity is increased above the nominal, such a speed is reached that heating becomes critical if the retrorockets are fired near perigee. As the speed is further increased, this area becomes larger and another critical area develops near apogee. At this point the reentry loads become too high if retrorockets are fired in the region of apogee. Finally, of course, as the speed increases, this danger area will cover the entire orbit. Therefore, for overspeed orbits, perigee and apogee are the points where retrofiring should be avoided. Actually, it is very unlikely that the Atlas would ever achieve an overspeed orbit because of the reliability of its guidance system; however, it is necessary to account for this eventuality. Again it is important to have real-time computing in order to tell where these areas are so that if an overspeed orbit is obtained, the retrorockets may be fired in a safe region.

Two maximum-energy curves are shown in figure 4: one where reentry is not possible at all and one where reentry is possible but is not safe from all points. Between the two maximum curves, the point of retrofire must be selected based upon the orbital conditions. The Mercury computing program provides means for selecting satisfactory retrofire conditions.

From the operational experience with Mercury, therefore, there are several conclusions which are believed to apply equally well to Apollo. First of all, flexibility is a must. Trajectories never seem to be finalized until the vehicle is launched. Therefore, the basic trajectory must be chosen so that consideration can be given to changes in spacecraft weight and in launch-vehicle and spacecraft performance. Another important factor which has not been a considerable influence in Mercury but which will play a very important part in Apollo is the month, day, and time of launch. The month and day of launch should be considered a random variable and the time of launch for a manned vehicle may vary as much as 4 hours. Also, from experience with Project Mercury, it has been found that the real-time computing based on the most accurate source of tracking velocity and position data has proved to be extremely valuable.

MISSION OPERATIONAL ANALYSIS

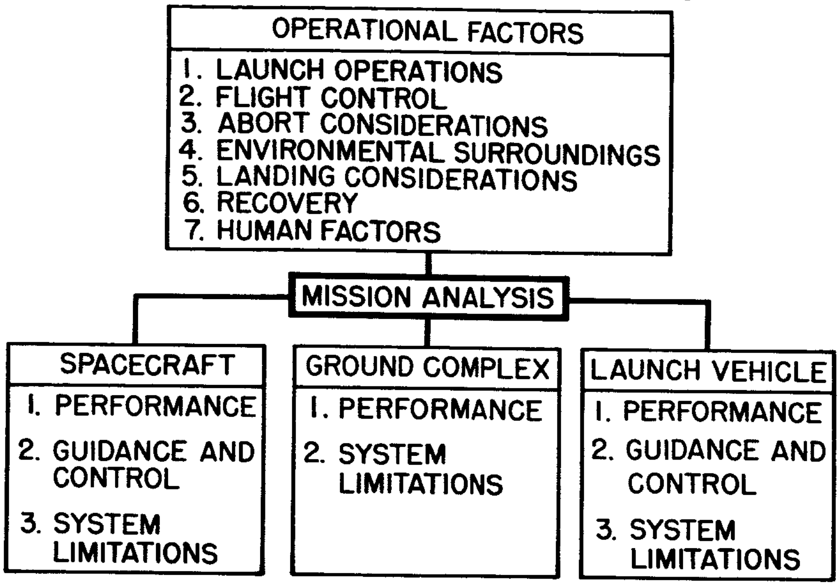


Figure 1

SELECTION OF ORBITAL INSERTION CONDITIONS

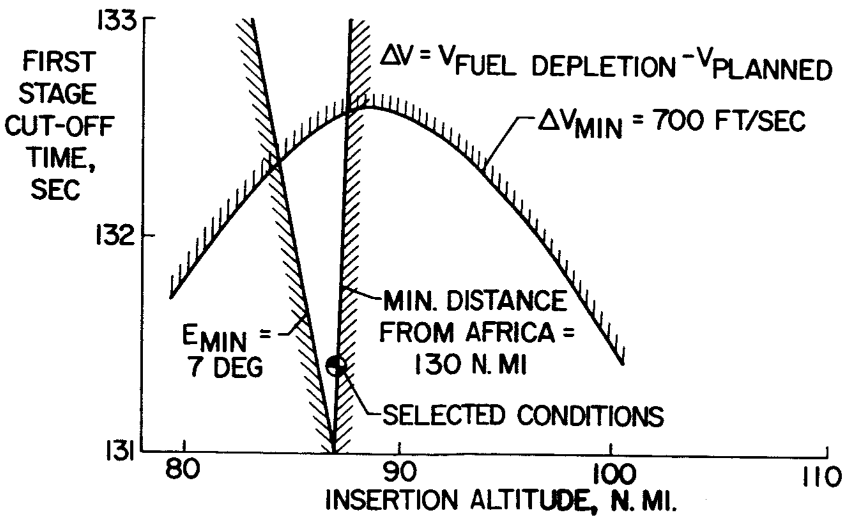


Figure 2

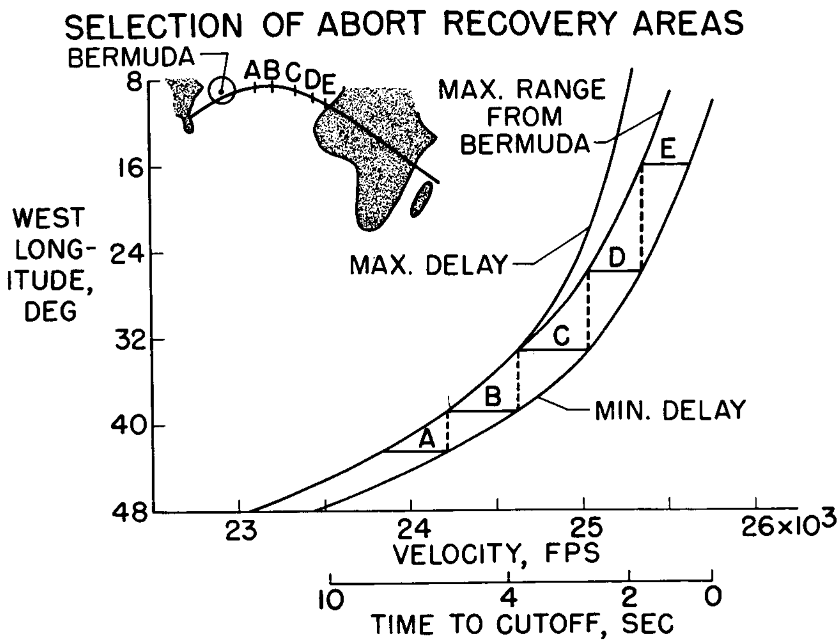


Figure 3

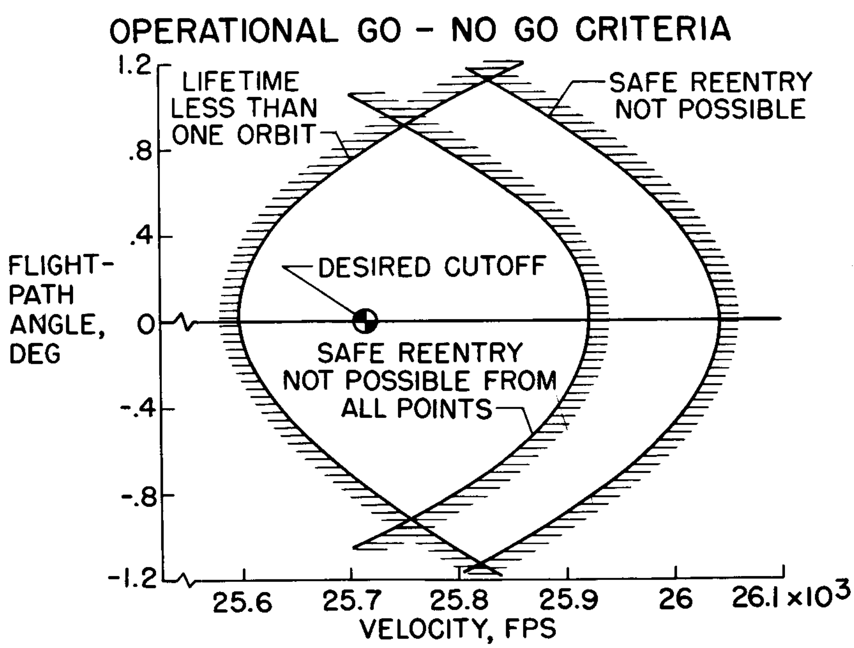


Figure 4

MERCURY PRELAUNCH OPERATIONS

By G. Merritt Preston and Dugald O. Black

NASA Space Task Group

INTRODUCTION

The preflight operations for Project Mercury are managed by the National Aeronautics and Space Administration. The organizational units involved in these preflight operations are shown in figure 1. Launch-vehicle operations are conducted by the U.S. Air Force in conjunction with General Dynamics/Astronautics for the Atlas launch vehicle and by the NASA Launch Operations Directorate of the George C. Marshall Space Flight Center for the Redstone launch vehicle. Launch-vehicle operations and organizations are the same as those used in developing the Atlas and Redstone weapon systems. It has been the intent of the NASA to use the past experience in this field as much as possible.

Range support is provided by the Air Force Atlantic Missile Range.

Spacecraft operations are conducted jointly by the McDonnell Aircraft Corporation and the NASA. NASA establishes the nature and scope of the testing from which McDonnell prepares detailed test procedures. These test procedures are then used to conduct the various systems tests. During these tests it has been the practice of NASA to assist in testing rather than observe so that the optimum use of manpower and the optimum monitoring of these tests can be obtained.

Spacecraft instrumentation was provided by the McDonnell Aircraft Corporation as part of the specification spacecraft. This method of approach was necessary for McDonnell to integrate the instrumentation systems into the spacecraft. Upon delivery of the spacecraft from the factory to Cape Canaveral Missile Test Center, the Preflight Operations Division of NASA takes full responsibility for all data-obtaining instrumentation.

The spacecraft, launch-vehicle, and range operations are integrated by NASA Launch Coordination Offices.

Contract inspection is provided for the spacecraft by McDonnell and for the Atlas by General Dynamics/Astronautics. Government inspection is also provided by NASA for the spacecraft and by the Air Force for the Atlas. This double inspection assures the flightworthy condition of the vehicle.

Preflight operations for the spacecraft differ from the normal mode of operation of other missile operations at Cape Canaveral. Mercury operations follow more the pattern of the airplane approach than a missile approach. Personnel at the launch site have a detailed knowledge of the spacecraft as well as a detailed knowledge of the components in the spacecraft. This permits an "on-the-spot" analysis of troubles and "on-the-spot" design changes to rectify these troubles. With programs such as Mercury and Apollo, this concept is mandatory if adequate flight reliability is to be obtained with a reasonable schedule. This approach is implemented by the local personnel having the authority to make changes to the spacecraft as required. Essentially, any changes that alter the basic philosophy of design are coordinated with the Space Task Group and the McDonnell factory before these changes are made. Changes that do not alter the basic philosophy are given to the parent organizations after the changes have been made. This approach requires highly competent personnel in the field.

Another deviation from the normal missile operation is that all systems tests are conducted by systems engineers rather than technicians. These engineers can, on the spot, evaluate from an engineering point of view the acceptability of the system. These engineers then have an up-to-date picture of the performance of each of the systems and can readily recognize any incompatibilities between systems that are created by slight changes from the specification performance of the individual systems.

A typical day-to-day operational schedule for a spacecraft is shown in figure 2. Of course, the time allotted for each of these operations may vary somewhat, depending on the difficulties encountered with each spacecraft. The five general operations may be broken down into the following categories:

(1) Systems tests

(2) Spacecraft modifications

(3) Mechanical work

(4) Spacecraft repairs

(5) Spacecraft servicing

Items (2) to (5) are included in the work periods shown in figure 2 and represent approximately 60 percent of the total time used in preflight operations; however, the time required for each of the preflight operations is discussed briefly in the order presented in the list.

First are systems tests which require about 30 percent of the total time spent in preflight operations. The systems tests consist of functional checks of each system independently and in conjunction with other dependent systems. A detailed discussion of each system test is included in a subsequent section of this report. Following the check-out of the various systems, the spacecraft is returned to the "white room" for further modifications and/or repair work.

The second item is modifications to the spacecraft. Modifications to the spacecraft result from experience obtained from the previous flights which indicates a need for a change and modifications in the design concept of the spacecraft that were conceived after the spacecraft was delivered. Changes also are made to the spacecraft at Cape Canaveral because of lack of parts while the spacecraft is at the factory. The time required to perform these modifications is approximately 25 percent of the time the spacecraft spends at Cape Canaveral. Some of the typical modifications that have been made to the spacecraft at Cape Canaveral are changes in the sequential system to improve reliability or take advantage of experience obtained from previous flights and changes to update the spacecraft components with later designs. These design changes result from previous flight tests, ground tests, or operational experience. Equipment in the reaction-control system, environmental systems, and communication systems have particularly been subject to change. In addition, numerous changes have been made in the mechanical systems for the same reasons. As with all programs, the instrumentation is changed as information is gathered and the need for more information is indicated. Also, instrumentation is deleted when the need for the information has been satisfied.

The third category of work at Cape Canaveral is the normal mechanical work required to prepare the spacecraft for flight. This work takes about 25 percent of the time. Typical work in this category is the final assembly of the spacecraft, mechanical fits, installation of the impact bag, and the mechanical fitting of modified parts to the spacecraft and adapter sections.

The fourth category is repairs and replacements, which require approximately 5 percent of the time spent at Cape Canaveral. Typical repairs are replacement of faulty equipment, repairing broken or damaged wires, and working off discrepancy reports or squawks.

The fifth category, servicing the spacecraft, requires approximately 5 percent of the time spent at Cape Canaveral. Included in this area are servicing the spacecraft with helium and peroxide for the reaction control system and installation of pyrotechnic actuators, rockets, landing-system parachutes, and batteries.

SYSTEMS TESTS

The various systems tests performed in Hanger "S" prior to mating at the launch-area complex are discussed in the following order:

(1) Reaction control system

(2) Automatic stability and control system

(3) Communications system

(4) Instrumentation system

(5) Electrical and sequential system

(6) Environmental control system

Reaction Control System

For the reaction control system, the spacecraft is outfitted with two mechanically independent fuel-handling and thrust-chamber groups for complete automatic spacecraft stabilization and independent manual spacecraft stabilization. Added redundancy has been achieved by coupling the independent systems electrically so that manual control of the automatic system can be realized by limit switches and automatic rate-stabilization damping can be realized by using the manual fuel system and thrusters. There are 18 thrusters in all; 12 are in the automatic system (6 of high thrust and 6 of low thrust) and the remaining 6 are applied 2 per axis on the manual system. Basically the systems differ from all other peroxide systems in that the peroxide is contained fully pressurized, nonvented from the time of ground checkout throughout the flight.

A new spacecraft is received from the manufacturer with the H_2O_2 system fully installed and ready for the first tests. Since the system is nonvented, extreme control of contamination must be exercised at all times. This means that close control of the inert gases used in the system, concentration and stability of all peroxide used for test and flight, and of all ground-support equipment designed to handle and store these fluids for use in the spacecraft must be exercised. For this reason all H_2O_2 tests are made in the peroxide facility adjacent to the hanger. (See fig. 3.) Briefly, the first test includes the following steps:

(a) The fuel bladder is pressure checked with gas.

(b) Gas is flowed through the system and thrusters to check the functioning of the thruster valves.

(c) A gas pressure check of the peroxide and pressurization halves of each system is made with helium. For this test the thruster nozzles are plugged.

(d) The system is filled with 35-percent H_2O_2 for 24 hours to monitor decomposition pressure rise as well as to precondition the system for use with 90-percent H_2O_2. The amount of pressure buildup affords a gross measurement of system cleanliness.

(e) A hydrostatic check of each system with 35-percent H_2O_2 is made to determine overall system-liquid integrity, including the thruster valves with the thruster nozzles unplugged.

(f) A pressure check is made of the high-pressure gas storage and handling mechanisms for each system.

(g) A functional check is made of the pressurization regulators which control the system pressures. The regulator shut-off valves are also checked for internal leakage to assure that decomposition surveillances will not be invalidated by high-pressure leakage should the storage bottles be charged.

(h) A 24-hour surveillance with 90-percent H_2O_2 is made to determine flight compatability.

(i) A flight-configuration functional check is made by pressurization from the spacecraft system and a static firing of each of the thrusters. A calibration of each fuel-quantity measurement is also performed at this time.

Any system rework or trouble shooting, either during the previously outlined hangar checks or prior to spacecraft mate with the launch vehicle, must be accomplished under conditions that are as clean as possible in order to avoid invalidation of system peroxide compatibility. Where at all feasible, during rework, the system is fed a slight positive inert gas pressure when changing a component to assure that all debris is driven out of the system as the work progresses. If it is determined that the rework is of a large enough extent to affect the system overall function, the hangar tests are rerun.

Automatic Stabilization and Control System

The tests of the automatic stabilization and control system are conducted with the spacecraft in a dynamic fixture which can be rotated at constant rates in roll and pitch (fig. 4). Yaw dynamic tests are conducted by rolling 90° and pitching.

The spacecraft is connected to a checkout trailer (fig. 5) and telemetry trailer (fig. 6) via the ground complex (fig. 7). The sequence tester in the checkout trailer can command any sequence of flight events necessary to the test. The following trailer monitors events and gathers data for calibrations.

The automatic stabilization and control system for the ground-support equipment in the checkout trailer is used to provide commands and to monitor the system operation. The following tests are performed after switches are pre-positioned and power is applied:

(a) The gyros are precessed in steps as indicated by digital dials in the checkout trailer. Telemetry uses these steps to calibrate the gyros. The spacecraft orbit mode is tested by recording the pulses and measuring them at 3°, 4.25°, 5.5°, 7°, and 8.5°.

(b) The amplifier-calibrator (digital computer) mode switching relays are checked by introducing commands from the checkout trailer, simulating commands from the spacecraft sequence system, and are monitored by test panel lights.

(c) The amplifier-calibrator logic is checked with a go—no-go automatic tester which performs over 600 tests in less than 2 minutes. Stops indicate the defective logic circuitry.

(d) The horizon scanners are checked by using a horizon and sun simulator.

(e) Astronaut mode-selection switching is checked after switching by performing continuity checks.

(f) The fly-by wire mode is tested by moving the hand controller in three directional axes. Reaction-jet indicator lights are monitored for closing of switches at the proper angles of stick tilt.

(g) The attitude-gyro alinement with respect to vehicle axes is checked by rotating around free gyros and computing misalinement angles which include fixture to spacecraft, spacecraft to gyro case, gyro case to gyro gimbal, and small inputs of earth and drift rates.

(h) Dynamic-thrust logic is tested by moving the spacecraft at various rates and through various angles. The records are compared with phase plane diagrams of the designed thrusting and switching logic.

(i) The programed attitudes such as retrofire and reentry are tested by applying the command, rotating the spacecraft to that attitude, and observing no reaction-jet lights, indicating gyro null.

(j) Reentry detection is accomplished by pitching the spacecraft until the 0.05g accelerometer pulls in.

(k) The dynamic test for the rate stabilization control system consists of moving the spacecraft at prescribed rates and observing reaction on recorders. The rate stabilization control system is tested with a special test panel. Null voltages and voltage necessary to trigger the reaction jet circuit are recorded.

Communication Checkout Tests

Following installation of all communications equipment, a closed-loop test of all communication subsystems is conducted. Parameters measured include:

(a) For HF and UHF voice transmitters: frequency-modulation quality and power

(b) For HF and UHF voice receivers: frequency, sensitivity, AGC action, and audio distortion

(c) For command receivers: single- and multiple-channel sensitivity, audio distortion, telemetry-transmitter interference, and command-code functioning of sequential relay

(d) For telemetry transmitters: frequency, power output, and plate current

(e) For UHF recovery beacon: frequency, pulse width, pulse spacing, group spacing, and peak power

(f) For HF recovery beacon: frequency, power output, and modulation

(g) For S and C band beacon: receiver sensitivity and frequency, pulse width, power output and frequency, antenna power division, and pulse space coding

During final hangar simulated flight test the following communication tests are conducted (all tests are open loop, utilizing all spacecraft antennas):

(a) For transmitters: frequency and relative signal-strength levels and modulation quality

(b) For receivers: frequency, relative sensitivity, and modulation quality (command function of command receivers is also exercised)

(c) For UHF and HF beacons: frequency, relative signal strength, UHF pulse width, pulse spacing, group spacing, and HF presence of modulation

(d) For TM transmitters: relative signal strength, frequency, and direct and reflected power flow in antenna transmission line

(e) For S and C band beacons: relative receiver sensitivity, receiver frequency, transmitter relative-power output, and pulse width

If, for any reason, communication equipment is removed following complete antenna installation, the antenna systems are retested for voltage standing-wave ratio, impedance match and correction of telemeter and UHF power-amplifier antennas, and power division of S and C band antennas.

Instrumentation System

After receipt of the spacecraft at Cape Canaveral, an orderly checkout of the telemetry systems is conducted. The systems and subsystems are removed from the spacecraft and inspected for any evidence of physical damage resulting from handling, voltage transients, corrosive atmosphere, arcing, heat, or dirt.

The component parts are modified as required to meet specification requirements and to include the latest design changes, and then bench tested, temperature cycled, and calibrated. Typical testing routines vary with the components being tested, but generally include the following determinations:

(a) Susceptibility to line voltage changes; that is, slow voltage changes, voltage spikes, and superimposed alternating-current voltages

(b) Changes in performance levels with temperature; that is, variations in gain, static and dynamic linearity, residual noise, center frequency and bandwidth if applicable, and others

(c) Input and output impedance measurements and susceptibility to loading

The subsystems tests would include the following:

(a) Adjustment of subcarrier oscillator preemphasis and transmitter deviation

(b) Adjustment of commutated subcarrier oscillator to keep the sidebands within the IRIG channel allocation

(c) Adjustment of the subsystem reference and carrier voltages with the spacecraft load

Following component modifications and checks and subsystems tests, the equipment is assembled and systems tests are conducted with all components connected and operating on the bench. These tests include the following:

(a) Final test of subcarrier oscillator pre-emphasis and transmitter deviation adjustment

(b) System noise survey with all equipment operating, including cameras and radio frequency

(c) Camera and programer compatability

(d) System susceptibility to noise and voltage spikes in the line and to low-voltage condition on the line

(e) Final through-the-system calibration for many channels

After completion of the calibration and bench tests, all components are inspected and reinstalled in the spacecraft.

During spacecraft testing in the hangar, all the instruments are tested and many of the transducers are calibrated; for example, the static-pressure transducers and attitude gyros. During the sequential testing, all the event functions are verified through the system. These tests determine the ambient data values for each channel and finalize many channel calibrations.

Electrical and Sequential System Test

System tests are conducted to verify individual system operation independent of other system operation. This isolation allows functional verification without jeopardizing other systems which normally interact with the system being tested. All redundancies are verified during these system tests. The following tests are conducted on the electrical and sequential systems at Hangar "S":

Electrical system -
(1) Direct-current power source
(2) Alternating-current power source
(3) Alternating- and direct-current control and distribution

Sequential system -
(1) Launch sequence
(2) Orbit sequence
(3) Escape or abort sequence
(4) Recovery sequence

Environmental Control System

The Hangar tests for the environmental control system are divided into two areas, those in the white room and those in the altitude chamber, and will be discussed in that order.

<u>White-room checks</u>.- White-room checks consist of individual- and group-component testing. The purpose of these tests is to insure that individual components meet the design specifications. Prior to the white-room checks the following servicing is performed:

(a) Oxygen bottles (normal and emergency) -

(1) Spacecraft are serviced to 3,200 lb/sq in. with N_2.

(2) Spacecraft with 7,500 lb/sq in. bottles are serviced with N_2 outside the hangar. The high-pressure leakage check is also performed outside the hangar.

(b) The coolant-quantity indicating-system bottle is serviced to 550 lb/sq in. with N_2.

(c) Water-cooling tank is serviced with approximately 39 pounds (maximum capacity) of water. Nitrogen is used in the white-room checkouts to reduce the hazards that are present when oxygen is used.

(d) A new CO_2 and odor absorber is installed and a time-usage log is kept from installation to launch.

The following is a list of the white-room checks:

(a) A suit-circuit-leakage-rate check is performed. This is done without pressure suit and astronaut.

(b) A leakage check is performed on the cooling water tank and associated plumbing.

(c) A leakage check is performed on the breathing oxygen bottles. For spacecraft with 3,200 lb/sq in. bottles this test requires approximately 8 hours. Accurate leakage rates are determined after a period of temperature stabilization. Other tests are performed concurrently with this test. For spacecraft with 7,500 lb/sq in. bottles, this test requires a longer period of time than for the 3,200 lb/sq in. system, depending on the time required for temperature stabilization.

(d) An electrical function check is performed. This test is performed to check the sequencing and functioning of all environmental control system components that require electrical inputs.

(e) A suit circuit shutoff valve leakage check is performed. Upon actuation, this valve isolates the pressure suit from the normal mode portion of the suit circuit. Actuation occurs during the emergency mode and during landing.

(f) A water-flow-rate check is performed on the cabin and suit cooling systems.

(g) A suit-circuit negative-pressure-leakage check is performed.

(h) A suit-circuit vacuum-relief-valve check is performed. This valve relieves the negative pressure which builds up in the flow line when and if the snorkle ball seats and fails to fall away from the seat.

(i) A suit-circuit pressure-regulator check is performed. This is a demand-type regulator and the check is run to assure that the regulator admits oxygen to the suit circuit according to specification.

(j) A ventilation-flow-rate check is performed on the emergency system (emergency rate) and the suit fans (numbers 1 and 2).

(k) An oxygen-system-transfer check is performed. This check is performed to assure that proper crossover from normal to emergency bottle occurs upon specified depletion of the normal bottle.

(l) A spacecraft-cabin pressure-leakage check is performed.

Altitude-chamber checks.- After the individual environmental-control-system components have been validated in the white room, the spacecraft is moved to the altitude chamber to check out the system as a unit. The altitude-chamber tests consist of unmanned and manned runs. It is mandatory that the unmanned runs be completed successfully prior to manned operation. During the unmanned runs the system is operated in the normal mode, the emergency mode, and the cabin decompressed mode. A maximum altitude of 120,000 feet is achieved during this run. During the manned run the astronaut is inserted into the suit circuit, and the system is checked again in the normal, emergency, and cabin decompressed modes. Again the maximum altitude attained is 120,000 feet. Prior to the altitude-chamber runs the spacecraft bottles are serviced with oxygen and the cooling water tank is topped off. An additional suit-circuit-leakage and cabin-leakage check is performed. The suit circuit and cabin are purged with oxygen prior to ascent. A continuous monitor of environmental-system parameters (temperatures, total pressures, partial pressures, etc.) is maintained during manned and unmanned runs. Successful completion of the altitude-chamber runs concludes the environmental-control-system checkouts.

HANGAR SIMULATED FLIGHT

Following the individual system tests, the electrical and sequential systems are operated in conjunction with other systems in a hangar simulated flight. The hangar simulated flight is designed to approach flight conditions as much as is practical. The following guide lines have been established:

(1) Absolute minimum of ground-support equipment and cabling

(2) Squib simulators installed at all squib locations to verify that proper energy is delivered to squibs

(3) Recorder channels paralleling squib simulators to verify proper sequence times of squib firing and to insure that there are no "back door" circuits

(4) Spacecraft configuration to be functionally as close as possible to flight configuration, including mounting the spacecraft on the adapter and connecting all flight wiring

(5) Test conducted on internal power

(6) All spacecraft systems follow actual flight profiles as close as practicable. This test is the final Hangar "S" test, and is designed to verify that all spacecraft systems are in proper operating condition.

SPACECRAFT SERVICING

Weight and Center-of-Gravity Determination

Upon completion of the spacecraft-system checkouts in the Hangar "S" area, the spacecraft is basically in a ready-for-flight condition and contains nearly all of the system components. Therefore, just prior to transporting the spacecraft from Hangar "S" to the launch complex is the logical time to weigh the spacecraft.

At this time the spacecraft, with the antenna canister and the retropack installed, is placed in the optical alinement fixture (fig. 8) located in the weight and balance area of the hangar. This fixture holds the spacecraft in the vertical position. In this position the spacecraft is adjusted so that it is within 0.006 inch of true vertical. This accuracy is attained through the use of three highly accurate transits mounted on two very heavy tooling bars at right angles to each other. The transits line up target points on the fixture to verify the level of the spacecraft. Through the use of a standard aircraft weighing kit, the weight of the spacecraft is obtained. The three weighing kit load cells are placed under the spacecraft (120° around the periphery of the 74-inch-diameter ablation shield), the true vertical of the spacecraft is checked once again, and then the load-cell readings are taken to within ±0.2 pound. In order to doublecheck this weight and determine variations in load-cell readings, the three cells are rotated and a second reading is obtained. With this weight plus the weight of the components not installed as yet (these items are weighed individually on beam scales, and are referred to as "paper weights"), the weight of the spacecraft is attained along with the X- and Y-axes and center-of-gravity location.

At this time, or just prior to the initial spacecraft weighing, the escape tower with the escape rocket (without igniter) is placed in the balance-alinement fixture (fig. 9). This fixture is adjacent to the optical-alinement fixture so that the same transits and tooling bar assemblies can be used as are used for the optical-alinement fixture. The escape tower is positioned horizontally in the fixture and is supported at three locations, two at the aft ring and one at the forward end of the escape rocket itself. The escape tower is then placed in an accurately horizontal position through the use of the transits and is weighed two times with the use of the weighing kit and three load cells. These two readings furnish the total weight and Z-station longitudinal center of gravity of the escape tower.

The spacecraft, with the antenna canister and retropack installed, is then placed into the balance-alinement fixture in a horizontal position. It is supported at three locations, two diametrically opposite at the heat-shield end of the spacecraft and one under the recovery section (cylindrical portion) of the spacecraft. Two weighings are taken of the spacecraft in this horizontal position by using the transits for leveling and the load cells with the weighing kit. This procedure gives a complete spacecraft weight which can be compared with the weight obtained in the optical-alinement fixture. It also furnishes the weights required to calculate the Z-station of the center-of-gravity location.

The spacecraft is then placed in the optical-alinement fixture in the vertical position. The escape tower is attached to the spacecraft so that now the complete spacecraft assembly is in the flight-readiness condition with the antenna canister, retropack, and escape tower all installed (fig. 10). This procedure gives the weight of the total spacecraft and the "X and Y" center-of-gravity locations. The Z-station center of gravity is determined by adding the spacecraft and the escape-tower weighings in the balance-alinement fixture together. The weight is within 1/10 of a pound, and the center-of-gravity location is within 0.10 inch.

Rocket Alinement

Once the centers of gravity are computed, the rockets are installed and alined. However, previous to this rocket installation the rockets receive the following inspection:

Pyrotechnic receiving and inspection.- All Project Mercury ordinance is shipped directly from the vendor to Cape Canaveral. It is checked and stored in the range contractor's solid-propellant storage area. The first check is a receiving inspection which consists of checking the following items:

(a) Dirty and pitted electrical-contact points

(b) Scratches, metal chips, filing, solder, rust, or burns on any surface

(c) Bent pins or dirty connector walls

(d) Loose connections, badly soldered joints, frayed wire, and cracked or peeled insulation

(e) Presence of moisture, corrosion, or foreign matter

(f) Misalinements of parts and improper assembly of parts

(g) Mismatch of parts through which dirt or moisture may eat and cause improper operation.

Following this check, the electrically initiated devices are checked for proper resistance values of the bridges. These checks are made by the range contractor (with NASA and McDonnell inspection) by using a Leeds & Northrup galvanometer, which provides a digital readout to three decimal accuracy. The resistance ranges of the Project Mercury ordnance are from 1.0 ± 0.3 ohm for a high-value squib to 0.21 ± 0.03 ohm for a low-value squib. All squibs are dual circuit and dual bridge.

The solid-propellant motors are visually checked for dents, moisture, good igniter wells, and proper assembly. A pressure check of each motor is made to meet the following specifications:

(a) The escape motor is pressurized to 50 lb/sq in. with an allowable leak of 1 lb/sq in. in 2 minutes.

(b) Posigrade, retrograde, and tower-jettison motors are pressurized to 30 lb/sq in. with 1 lb/sq in. drop in 2 minutes allowed.

(c) The escape motor is disassembled and the grain is inspected by a vendor representative prior to the pressure check. The retrorockets are X-rayed prior to acceptance for case bonding.

All pyrotechnics are made into kits about 1 month prior to launch. All squibs are resistance checked before being included in the kits. The kits include one full set of ordnance and some spares. The internal pyrotechnics are installed in the spacecraft in Hangar "S"; however, no pyrotechnics are connected until stray voltage checks are made. After the ordnance is installed, an electrical check is made of the circuit including the bridge resistance. The spacecraft wiring resistance is determined prior to installation of pyrotechnics. No rocket-igniter circuits are checked in the hangar. After the spacecraft is delivered to the launch complex, the rocket igniters (except the escape rocket) are checked before the spacecraft is hoisted into position on top of the launch vehicle.

<u>Escape-rocket alinement</u>.- The objective is to aline the rocket-thrust line with a point off the center of gravity so that the escape tower with spacecraft will veer away from the launch-vehicle line of motion when separated. The center of gravity for this mode (spacecraft with tower) is around $Z = 165.44$ (this is in the vicinity of the recovery compartment; for reference, the rim of the heat shield is $Z = 104.50$). The rocket manufacturer marks the center point of thrust on the base of the rocket.

With the location of the spacecraft center of gravity and the center point of thrust on the rocket, the tower is shimmed at the base so that the line of thrust passes through a point where $X = 0.520$ inch and Y = 0.300 inch away from the center of gravity at the Z = 104.50-inch plane in the same quadrant.

This is accomplished to within 0.10 inch by triangulation with the optical-alinement fixture.

Retrorocket alinement.- The objective is to have the thrust line of each rocket pass through a point not more than 0.08 inch from the spacecraft center of gravity which is usually located somewhere close to a point at Z = 122.00 inches, X = 0.25 inch, and Y = 0.25 inch off center. This objective is also accomplished by triangulation and the use of the optical equipment on the optical-alinement fixture.

When the location of the spacecraft center of gravity is known, each of the retrorockets is adjusted so that the physical center of each nozzle falls in line with the center-of-gravity point. This line is then extended optically to the floor to locate a third point on the line of thrust. The next step performed by use of a special tool is to aline the plane of each nozzle throat at 90° to the thrust line. To do this the base of the tool is placed over the thrust point on the floor just determined and the top of the tool is fitted into the nozzle. The rocket is then shifted in the retropack until the nozzle center line is concentric with the tool center line.

LAUNCH COMPLEX TESTING

The tests conducted at the launch complex are shown in figures 11 and 12 for the Atlas and Redstone complexes, respectively. Prior to mating the spacecraft to the launch vehicle, the complex wiring is checked with an automatic checker to assure that all electrical lines have continuity and proper end points.

As shown in figures 11 and 12, the first operation at the complex is the mechanical mate of the spacecraft to the launch-vehicle adapter. This includes setting the spacecraft on the adapter, installing the clamp ring, and hooking up all electrical, hydraulic, and gas lines. Also, at this point, the escape tower is checked for fit with the spacecraft and complex and is then removed for refit at a later date. This procedure is used to avoid having the escape rocket with the destructive potential in the area any longer than absolutely necessary. During the mechanical mate all ground service equipment is moved into place and connected.

From this point the order of testing on the two complexes varies somewhat due to differences in launch-vehicle preparation and handling; however, the tests are similar. For example, the overall tests 1, 3, 4, and 5 shown in figure 12 are essentially the same as the special interference tests shown in figure 11. This is also true for the electrical mate and overall test 2 shown in figure 12 which are comparable with the abort and interface checks in figure 11. Part I of the radio-frequency compatibility shown in figure 12 is similar to part I of the flight acceptance composite tests of figure 11; also, the simulated flight of figure 12 is similar to part II of the flight acceptance composite tests of figure 11. The overall test 6 of figure 12 is the same as the radio-frequency abort test of figure 11. The other tests of similar names shown in figures 11 and 12 are essentially the same.

In order to simplify the discussion of complex testing, the discussion of the tests is continued in the order presented in figure 11. The mechanical mate is followed by the spacecraft-systems tests in which power is applied to the spacecraft from an external source and all systems within the spacecraft are operated and checked out. During this testing, end-to-end checks of calibrations are made. After all systems have been checked satisfactorily on external power, they are switched to internal power and a brief check is made of all systems while operating on internal power.

Next are the interface and abort tests in which a check is made of the various paths by which information is transmitted through the spacecraft—launch-vehicle interfaces as well as the hard-wire paths by which aborts are transmitted to the spacecraft.

Next is part I of the flight acceptance composite test. This test is designed to prove that the spacecraft radio-frequency systems are compatible with the launch vehicle and range and is performed with all systems (launch vehicle, spacecraft, and range) operating to verify compatibility and no interference.

Part II of the flight acceptance composite test not only permits a check of radio-frequency systems compatability with the service structure moved away but also contains a mock count which permits the launch crew to operate under simulated launch conditions.

Next is a series of special interference tests. In these, all launch-vehicle and spacecraft equipment is turned on, and the spacecraft and launch-vehicle systems are operated through all flight phases to determine whether any stray circuits created by either the spacecraft or launch-vehicle equipment can adversely affect the launch-vehicle or spacecraft functions. Also included in these tests is a check of the various redundant paths for sending and receiving radio-frequency abort signals.

Following the preceding tests, there are 2 days in which the reaction control system is checked out by using 90-percent H_2O_2, the spacecraft is serviced and made ready for launch, and the pyrotechnic igniter devices aboard the spacecraft are checked for continuity. In the reaction-control-system checks the system is filled with 90-percent H_2O_2, and after a pressure surveillance of approximately 12 hours the reaction nozzles are fired with the spacecraft circuitry. The pyrotechnic checks consist of installing all pyrotechnics not already in, such as the escape tower and tower-clamp ring bolts, and checking out the initiator circuits. During the checks all pyrotechnics except the escape-rocket igniter are installed.

The spacecraft servicing in preparation for launch consists of such items as stowage of the survival gear, food, servicing all recording devices with fresh film or tape, and topping off O_2 bottles.

The final test is the launch countdown which requires 2 days. The first half of the countdown consists of a brief check of all systems on both external and internal power. Then, a power-on stray voltage check was made followed by a power-off voltage check and an electrical-connection check of all pyrotechnics except the escape-rocket igniter. Next, the H_2O_2 system is filled in preparation for the launch, the escape rocket igniter is installed (electrical hook-up of this igniter is done later as part of the count), and cameras and tape recorders are serviced as required.

The second half of the countdown consists of a further check of compatibility of the launch-vehicle and spacecraft systems, insertion of the payload, loading of launch-vehicle fuel and lox, removing the service structure, powering up the spacecraft and launch vehicle, observing telemetered signals from all systems, switching to internal power, and launch.

PROBLEM AREAS

During the preflight tests of the various systems, many problem areas developed. These problem areas are discussed subsequently for each of the systems.

Reaction Control System

The problems in the reaction control system and the method of solving them are as follows:

(1) The first problem is contamination which is generally in the forms of nonsoluble aluminum oxides, believed to be caused by moisture

remaining in the system, and metallic chips or particles, believed to be due to overtorquing of joints. Such contamination results in two major problems:

(a) Excessive decomposition of the 90-percent H_2O_2

(b) Unreliability of systems components such as sticking valves or leaks

Contamination has been partially controlled by strict adherence to environmental working cleanliness, quality control of components including H_2O_2, vacuum drying of H_2O_2 system after use, and the use of flare seals to lower torque requirements for sealing joints.

(2) Several throttling valves have been rejected due to excessive actuating force or leakage around the valve actuating shaft.

To correct these problems, the body cap surrounding the steel actuating shaft was stripped and hardcoated to reduce the possibility of galvanic corrosion. This resulted in an increase of up to 0.003 inch in the diameter of this hole without exceeding seal design tolerances. However, this increase in diameter allowed the O-ring in the cap body to extrude under pressure, resulting in high actuating force or leakage and in some cases the O-ring was cut by the sharp, hardcoated edge adjacent to the O-ring groove. The addition of a Teflon backup ring in the O-ring groove has eliminated this problem.

(3) Seals in quick-disconnecting valves and couplings have been unreliable and have been replaced with manual shutoff valves.

Automatic Stabilization and Control System

The following problems of the automatic stabilization and control system together with their methods of correction are presented as follows:

(1) The spacecraft lost corrective thrust if attitude error exceeded 8.5°. This loss was corrected by adding a switch to the amplifier calibrator (a digital computer) to provide for switching from the orbit mode, which uses short, low-impulse pulses for attitude control only, to the orientation mode, which uses continuous thrust commanded by attitude rate. This switching takes place when the spacecraft gets outside the bounds set by the retrofire interlock circuit. This change was later permanently wired in.

(2) The spacecraft was capable of only one pitch attitude while in orbit. By adding a switch to provide for either -14.5° or -34° orbit attitude, the optimum position could be achieved for orbiting.

(3) Slaving relays in the yaw loop were switched at the input when slaving was not desired; the noise created in the following circuits caused erratic yaw slaving. This was corrected by changing the switching to the output side.

(4) The starting capability of the horizon scanners did not prove to be 100-percent reliable. A change was made to run these continuously with only the output being programed.

(5) Communications transmitters interfered with transmission of slaving and ignored voltages from the horizon scanners. The scanners are located in the antenna canister. The correction consisted of adding bypass capacitors on the alternating-current leads to the motors and wrapping all wires with aluminized tape.

(6) The retrofire permission interlock had an arming circuit which required a retrograde attitude signal and a signal provided by the spacecraft changing attitude toward the proper pitch attitude by about 19°. When the retrograde attitude was changed from -14.5° to -34°, there was not enough travel to cause the circuit to operate. This was corrected by opening the output.

(7) A signal is given to the system 5 seconds after the sustainer-engine separation which activates the orientation mode and allows the spacecraft to rotate in yaw 180° and pitch down to -34°. The sequence wiring which commands this was not redundant. A failure in this single chain would cause spacecraft loss in an unmanned flight. The system was made redundant. Additionally, the calibrator circuit which provides this output is parallel when the retrofire signal is given, assuring spacecraft turnaround and pitch down during emergency abort from orbit.

(8) Formerly a retrograde attitude signal was held in for the first 5 minutes of free flight. At the end of this time the spacecraft reoriented to -14.5° orbital pitch attitude. Since the orbital attitude was changed to -34°, a considerable fuel saving was accomplished by removing the command and allowing the vehicle to go into the orbit mode as soon as it becomes stabilized.

(9) Manual control was refined and made more redundant by the addition of a rate-stabilization-control system. This was done by using potentiometers on the control stick, the voltages from which are summed with outputs from the rate transducers to operate solenoids by

using the manual system fuel. The system is also used for roll spin during reentry if the astronaut elects.

(10) When the slaving circuitry in the manual test panel was switched out, a 10K resistance to ground appeared that would not allow proper horizon-scanner scale-factor adjustment. The switching was modified to correct this resistance.

(11) Horizon-scanner-ignore signals could not be observed. Circuitry and indicator lamps were added to the checkout equipment.

(12) During dynamic tests orbital-pitch slaving could not be removed. A test switch was added to provide this.

(13) The attitude gyros could not be caged from the blockhouse. This feature was added.

(14) Since there was no special test for the umbilical to the spacecraft, test kluge cables were provided to isolate circuits during trouble shooting. Addition of these made all necessary test points available.

Communications Systems

The problems in the communication systems and the solution of these problems are presented as follows:

(1) It was found that single-pulse beacons can be interrogated by any radar which is within range and which is adjusted to the beacon receiver frequency. This permitted beacon interrogation by a large number of non-Mercury radars either accidentally or intentionally. Use of a double-pulse interrogation system provides for a coded system which reduces the probability of undesired interrogations. The increased security against undesired interrogations is obtained at the expense of somewhat reduced reliability.

(2) Whenever the original Mercury beacon is interrogated by two radars so that the second radar's pulse occurs within the beacon recovery period after the first radar's pulse, the result is the complete blanking out of beacon-reply function for both pulses. Beacons (C band) are now being modified with a beacon-receiver blanking gate which prevents any reaction to the second radar's pulse but in no way affects the reply from the first radar's pulse.

(3) The present specification range for C and S band beacon delays is 0.4 to 1.6 microseconds. This range allows the return pulse

for some beacons to fall within the skin tracking gate of the tracking radar. The result is an unstable oscillation of the tracking radar between skin and beacon-return operating modes. This condition can result in loss of track. At present, beacons are not being modified to change this delay range. Considerable difference of opinion exists as to the seriousness of this problem.

(4) Multiple antenna elements around the periphery of the spacecraft produce numerous deep nulls between the elements. This affects the angular and range tracking accuracy of the ground radars to an extent which makes orbit-prediction calculation questionable. This nulling effect appears to be reduced by inserting sinusoidally varying phase shift at an appropriate audio frequency into one of the antenna transmission lines. Ferrite phase shifters have been tested and improvement has been demonstrated.

(5) Severe audio-frequency noise has been encountered in both received and transmitted intelligence. Tests have indicated that the 400-cycle inverters are the source. The noise gets into the communications system in two ways: coupling between audio and alternating-current power wiring, and magnetic field induction between inverter and microphone. Wiring routing changes and incorporation of an Electro Voice microphone have demonstrated reduction to a satisfactory level.

(6) High-frequency communication checkout with the Mercury Control Center from the launch pad has been unsatisfactory. In order to protect the transmitter from antenna mismatch, an isolation attenuator must be installed during tests. Antenna mismatch is inherent because the spacecraft when attached to the launch-vehicle represents a configuration considerably different from that in flight. A suggested test method is to feed the transmitter directly to a matched gantry mounted high-frequency antenna. This would provide spacecraft to Mercury Control Center communications but would not check out the multiplexer and spacecraft antenna. Basically, there does not appear to be any method of simulating spacecraft high-frequency antenna flight configuration on the launch pad.

Instrumentation System

There have been numerous problems in the instrumentation system which have not been amenable to corrective devices without a systems redesign such as stacking of components, cable mismating, and inflexibility of design; however, there have also been numerous problems which have been adjusted at least in part. These problems are listed as follows, together with the correction:

(1) There have been many corrections to the commutators such as replacement of internal power supply, redesign of amplifiers, and segment-by-segment calibrations; however, none have been completely satisfactory and the unit is being redesigned for manned orbital spacecrafts.

(2) Telemetry transmitters have been plagued with power output and frequency troubles and to date the only solution has been to examine carefully all transmitters and select the best of those available. In addition, the transmitters are extremely sensitive to temperature due to the use of germanium transisters. This condition has been partially overcome by mounting the transmitters on the cooling water tanks.

(3) For camera failure, also, the only solution to date has been to select the best of the cameras available by means of exhaustive pre-installation checks.

(4) Electrocardiogram amplifiers had to be redesigned in order to reduce line-voltage sensitivity and to improve stability. New power input filter regulators of improved design were installed, and the feedback system was redesigned to eliminate a tendency to oscillate and to improve common mode rejection.

(5) The resistance pickups are being replaced with thermocouples to eliminate sensor failure and inadequate sensitivity. The O_2 partial-pressure pickup has been redesigned to provide a case material which is not sensitive to oxygen atmosphere. Stick-motion sensors are not being replaced; however, very special adjustment is required for even gross data. Respiration-rate and depth-sensor sensitivity has been improved by moving them to the upper lip of the astronaut. The voltage sensor has been redesigned to bias out low potentials and, thus, increased the sensitivity.

(6) It was found that the calibration curves supplied by the vendor were inadequate and that in some cases no provisions were made for field adjustments. Procedures were developed by NASA for field adjustment, and all calibrations are made during bench and systems tests.

Electrical and Sequential System

Problems in the electrical and sequential system are defined and corrective procedures are outlined as follows:

(1) Because of fuse failure, fuses were replaced with solid conductors in any circuit in which a fuse failure could cause catastrophic result. Fuses in series with 1-ohm resistors are being added to the

squib firing circuits to reduce the magnitude of the line pulse on the bus caused by firing the squibs.

(2) The inverters were found to be sensitive to the following conditions:

(a) Perturbations of main bus voltage

(b) Output coupling (alternating current)

(c) Low voltage at input during switching

Any of these conditions or combinations of them were found to cause the inverters to stall. In other words, the input current would increase to approximately 70 amps and the output would drop to zero. Filters were added to minimize the inverter sensitivity to line surges. Wiring was changed so that output coupling could not occur. Switching procedures were changed to hold inverter switching to a minimum. Inverter heating was also found to be a problem; at inverter case temperatures of approximately 200° (depending on the inverter), the inverter will stall and blow the fuse. This does not seem to cause any damage to the inverter and once it is cooled it will again operate properly. Heat sinks have been added to alleviate this problem.

(3) The general Mercury design philosophy was to utilize the astronaut as the backup or redundant control path for many of the sequential functions. Since many of the first Mercury flights were unmanned, it became apparent that provisions would have to be made to design in automatic redundant control paths for functions normally performed by the astronaut. In this manner, total vehicle reliability was preserved. Several examples of astronaut backup functions which were made automatic for unmanned shots are as follows:

(a) Spacecraft-separation bolt emergency control.

(b) Tower-separation bolt emergency control.

(c) Attitude permission bypass (ground control)

(d) Retrorocket-fire emergency control

(e) Drogue-chute-deploy emergency control.

(4) The spacecraft-adapter and tower-clamp-ring limit switches actuated prematurely in Little Joe flights. This problem was corrected by the addition of arming circuitry which makes the limit switches functionless until they are required.

(5) The sensitivity of timing devices to line surges was alleviated by adding a capacitor across the input of each timer. A 1-ohm resistor was also added in series with squib fuses as mentioned above for line-surge supression.

(6) A low-voltage problem has existed on all unmanned flights to date. This problem is alleviated somewhat by shallow cycling (slight discharge and recharge) at the pad approximately 3 days before launch. Also, battery-on time before flight is held to a minimum. On later spacecrafts there is a proposal to utilize the cells of one of the standby batteries to boost the main bus voltage. This should raise the main bus voltage 2 to 3 volts.

(7) Original testing plans did not provide for any end-to-end testing. This was corrected by the addition of the Service Engineering Dept. Report 77 which checks spacecraft wiring right up to the time that the squib disconnects with all spacecraft systems operative.

Environmental System

Problems in the environmental system and their solutions are presented as follows:

(1) The high-pressure shutoff valves tend to leak in the full-open position due to an eccentric loading on the valve shaft. The opening procedures have been revised to back the valve off one turn from full open. No leakage problems have developed in this position.

(2) The original sponge used to collect the condensate from the suit circuit was hard and somewhat brittle in the dry state. This created a problem in that it was necessary to saturate the sponge before operation. A new type of vinyl sponge, which is soft in the dry state, was incorporated; thus, this problem was alleviated. The separator piston position was difficult to determine and had a tendency to stick in the compressed position. In order to solve this problem, a new piston seal was incorporated and a magnet was attached to the piston for externally determining the position.

(3) The majority of cleanliness problems have stemmed from the fact that adequate cleaning facilities have not been available at Cape Canaveral. Rigid cleaning specifications have been set up for the existing facilities at Cape Canaveral. However, an interim solution to this problem is to contract out the required cleaning which cannot be obtained at Cape Canaveral and to allow the remainder to be cleaned at Cape Canaveral facilities, all with close monitoring by inspection. Plans are presently being formulated by Pan American World Airways, Inc.,

to construct a new cleaning facility at Cape Canaveral which should handle all cleaning requirements for Mercury and subsequent programs.

(4) During altitude-chamber operations, freon vapors used to cool the spacecraft were found to contaminate the chamber atmosphere. A solution to this problem has been to route a bypass suction line to the chamber bottom. Due to the high density, all freon will settle to the bottom of the chamber and be exhausted. The temperature of the breathing oxygen for the observer was found to be uncomfortably high. A cooler was installed in the oxygen supply line to lower the incoming oxygen temperature. Lack of confidence in the strength of the original single-pane chamber windows resulted in a window redesign. The redesign consisted of increasing the glass thickness, together with lamination. Also, a safety blowout-proof pane was installed outside each window.

Mechanical Systems

Problems of mechanical systems and their solutions are presented as follows:

(1) The periscope had two major problems:

(a) Failure in motor drive

(b) Failure of periscope door to close fully

The drive motor was replaced with a larger motor with no subsequent failures. An analysis indicated the periscope door closing link could be stressed for a larger loading than that called for in the rigging specifications and, after doing this, and adjusting the limit switches, there were no further problems with door latching.

(2) It was found that the combination of impact and wave action destroyed the impact skirt and retaining straps with a resultant loss of the heat shield. This was corrected by adding cables having swivel ends.

(3) The time required for fitting adapter to launch vehicle has been excessive.

(4) Interference of the tower with antenna canister and warping of the clamp ring are the two most serious problems encountered in fitting the escape tower. The interference of tower with antenna canister is due to the extremely close fit and, with the sheet-metal tolerances used in manufacturing, this will remain a problem and will

require trimming of both tower and canister for each spacecraft. The warping of the clamp ring was corrected by redesign and making the clamp ring and aerodynamic fairing one complete unit.

(5) Due to the material of which the impact skirt is fabricated (rubber-coated fiber glass), bending or creasing of impact skirt results in breaking of the fibers; therefore, it has been suggested that the skirt be changed just prior to final stowage. Time considerations have prevented this; however, the number of times that the skirt is folded for storage has been reduced to a minimum with satisfactory results.

(6) Battery vent lines are constantly being modified due to spacecraft configuration changes and while this is fairly easily done the manifold lines are not adaptable to rerouting.

Operational

One operational problem is the inaccessibility of equipment due to the necessity of stacking of components within the limited space afforded by the spacecraft. This stacking of equipment makes it necessary to move several wire bundles and to remove several pieces of equipment to install the batteries. Since it is desired to have the spacecraft in flight configuration before the hanger simulated flight, the batteries must be installed before this test. This requires that the batteries be activated approximately 1 month before flight. Having the batteries installed also leads to the difficulty that if any batteries should fail after installation, trouble would be encountered in replacing them. Fortunately, however, the service life of the batteries used in the Mercury spacecraft has been very good once they have been activated and installed in the spacecraft. Figure 2 shows the following additional effects of stacking of the components:

(1) Approximately 60 percent of the time the spacecraft is at Cape Canaveral is spent in performing work on the spacecraft. This long period of time is caused by the difficulty of working on any of the components in the spacecraft.

(2) In the process of removing equipment from such cramped quarters, wires are frequently broken necessitating repair and recheck.

(3) Installation of connector plugs in such cramped quarters frequently results in wrong connections. This could be corrected by a new design by making all adjacent connectors of different design.

(4) The large number of pyrotechnic devices used in the Mercury configuration require a safe system for shorting the electrically

initiated items. The Mercury design is such that it is necessary to separate wiring and install shorting plugs during firing checks and then only a continuity check can be made after reconnecting the cables. There is no better solution known for the present design.

Another problem is the inflexibility of the systems in that any change in a system usually means cutting and splicing electrical wires to effect a change.

The present concept of Mercury spacecraft checkout of individual systems requires a separation of plugs within the spacecraft in order to install closed-loop connections between the spacecraft and checkout trailer. This trailer is used to simulate the signal inputs which the spacecraft systems would receive during a flight. This separation of plugs within the spacecraft makes it impossible to perform a complete end-to-end check of the spacecraft's equipment with all plugs connected as they would be during flight. It is suggested that future designs incorporate check points which will not require separation of spacecraft wiring. The checkout trailer is limited in space, and for tests such as simulated flight where all spacecraft systems are operated, it becomes extremely crowded. These crowded conditions are conducive to error and accidental tripping of unguarded switches.

CONCLUDING REMARKS

The following points resulting from the experience with Project Mercury that might be applicable and helpful in planning for Project Apollo should be emphasized. However, it should be stressed that these remarks are not made in a derogatory manner concerning Project Mercury. Many of the conditions that exist resulted from the necessity of keeping the spacecraft size and weight to a minimum and from the fact that this is the first U.S. venture into manned space flight.

It should be stressed, first, that it is desirable for the Apollo to be a serviceable design - one that is designed to operate, one that is designed to be modified and kept current, one that is designed with accessibility, one that is designed with durability and equipment that will survive the normal day-to-day handling. Since the Apollo vehicle is considerably larger than the Mercury vehicle, additional space should be available for installing the equipment so that these pieces of equipment can be removed without disrupting wire bundles or other pieces of equipment. Along this same line, considerable equipment is available from Mercury and other missile projects; it is hoped that off-the-shelf equipment, particularly from Mercury will be used in all cases in which it is practicable. This trend not only improves logistics of supply but increases the reliability through previous knowledge of equipment, equipment idiosyncrasies, and the methods of checkout of the equipment.

Next, considerable study should be made of the union of solid-state electronics with what is called mechanical switching devices such as relays, switches, and so forth. Problems in this area have harassed all people in the space age.

One lesson learned from Project Mercury was that modifications of the space vehicle in the field must be made in the interest of reliability and safety. An atmosphere should be created in which proper changes are encouraged. Since these vehicles must have a mission reliability approaching 100 percent, experience from previous flights must be incorporated before subsequent flights are made. In addition, as the design evolves, new and better methods of design will be conceived. When these methods materially improve the reliability and safety of the vehicle, they should be incorporated. One of the fortunate circumstances of Mercury has been that a good portion of the crew that designed the spacecraft at McDonnell followed the spacecraft to Cape Canaveral to perform the preflight checkouts. This crew is now in residence at Cape Canaveral and has assisted the NASA personnel in preparing the spacecraft for flight. Typical of the caliber of people is the McDonnell Design Project Engineer who is now the McDonnell Base Manager. The presence of this type of personnel at the preflight operations station has made it possible for design changes to be made on Mercury with a minimum of effort since these same personnel know all the compromises that have previously been made in obtaining the original design. They also are familiar with the design of the spacecraft so that these modifications can be made with a minimum of time. One final point which should be made is that Project Apollo will be a research and development operation and can best be compared with Project Mercury and the research airplane-type operation conducted at Edwards. As far as possible reuse of the spacecraft should be considered. Provisions, therefore, should be made to ensure that the equipment is ready for reuse with the minimum of recycling. The checkout of the Mercury vehicle requires a large amount of effort. This effort has proven worthwhile by the successful missions accomplished to date but means must be explored by the personnel of Project Apollo to reduce the effort required. This is not meant to imply automatic checkout equipment. Limited use of automatic equipment is desirable and is used on Mercury; however, caution should be exercised in widespread use of this type of equipment.

PRELAUNCH OPERATIONS

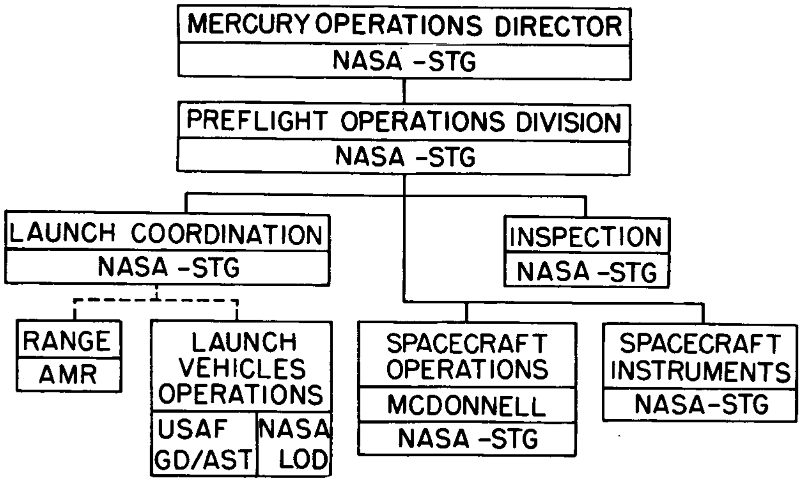


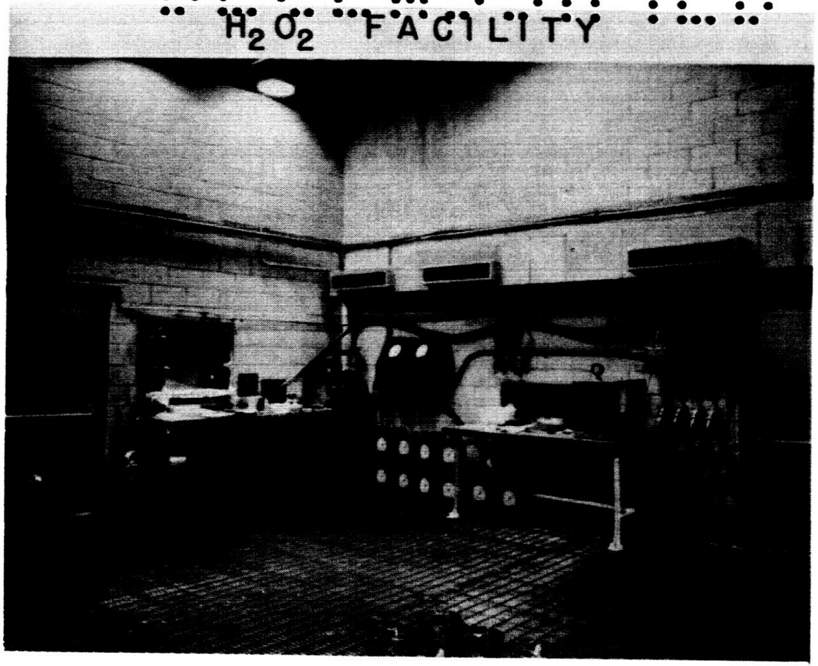
Figure 1

PREFLIGHT SPACECRAFT OPERATIONS

OPERATION	PERCENT TIME REQUIRED
• SYSTEMS TESTS	30
• SPACECRAFT MODIFICATIONS	25
• MECHANICAL WORK	25
• REPAIRS AND REPLACEMENTS	5
• SERVICING	5
• COMPLEX TESTING	10

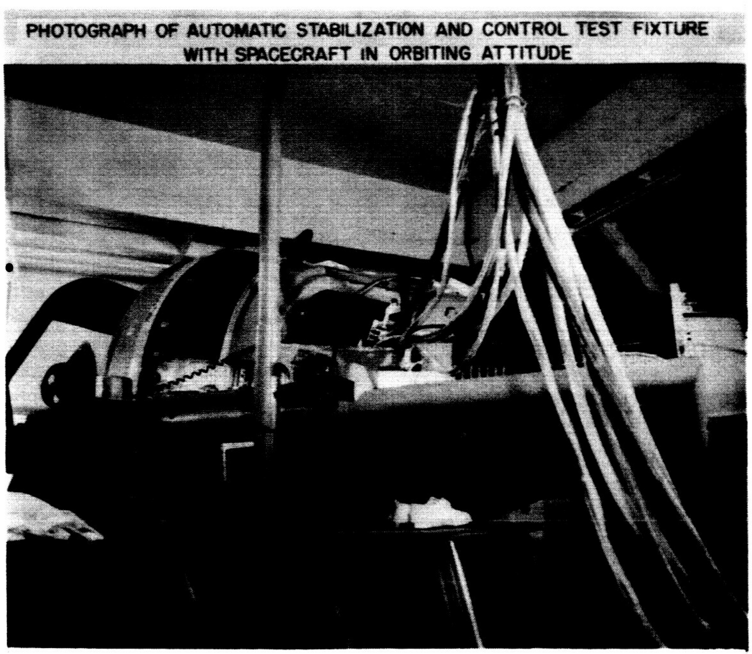
Figure 2

[redacted]



S-61-58

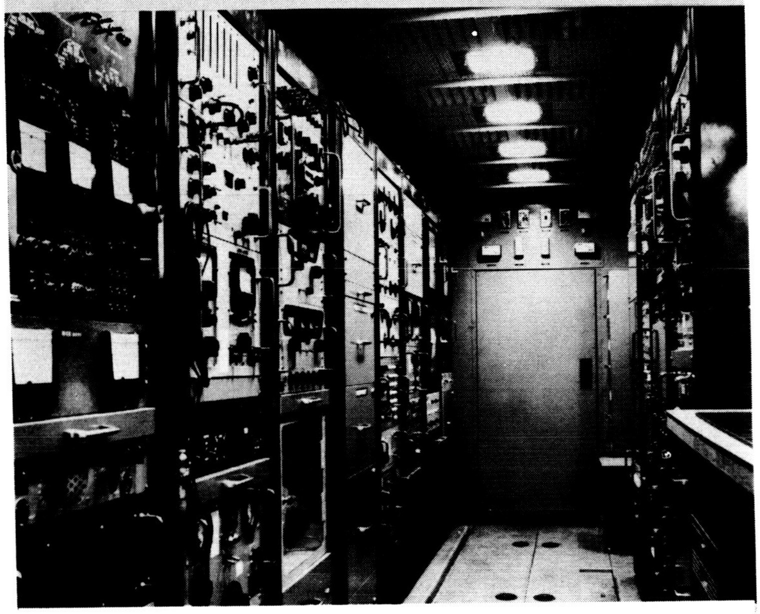
Figure 3



LOD-61-5082

Figure 4

INTERIOR VIEW OF CHECKOUT TRAILER USED FOR HANGAR "S" TESTS AND EARLY TESTS AT THE LAUNCH COMPLEX



LOD-61-5952

Figure 5

INTERIOR VIEW OF TELEMETRY TRAILER USED FOR SPACECRAFT TESTS

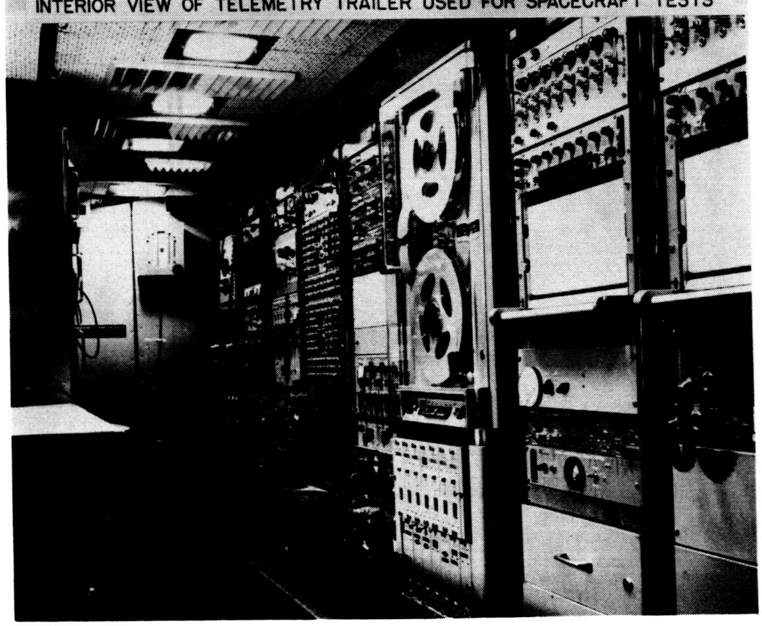


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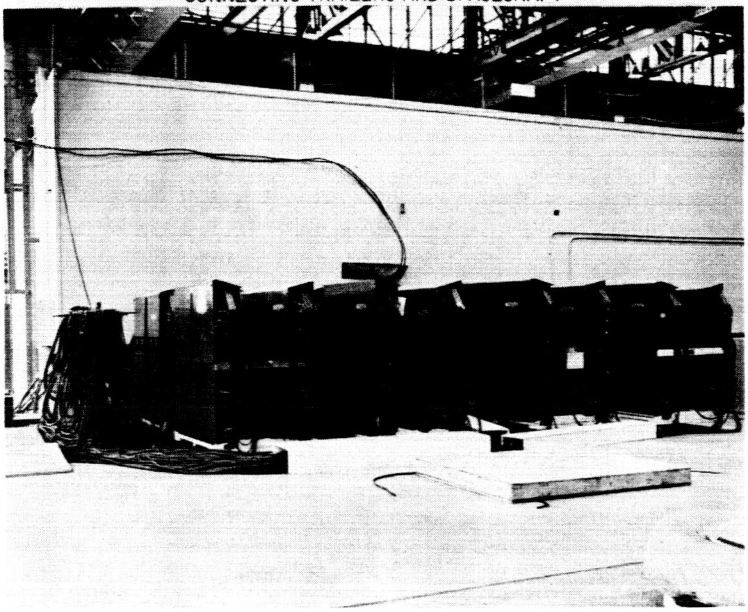
Figure 6

[redacted]

OVERALL VIEW OF TERMINAL BOARDS USED IN GROUND COMPLEX FOR CONNECTING TRAILERS AND SPACECRAFT



LOD-61-6051

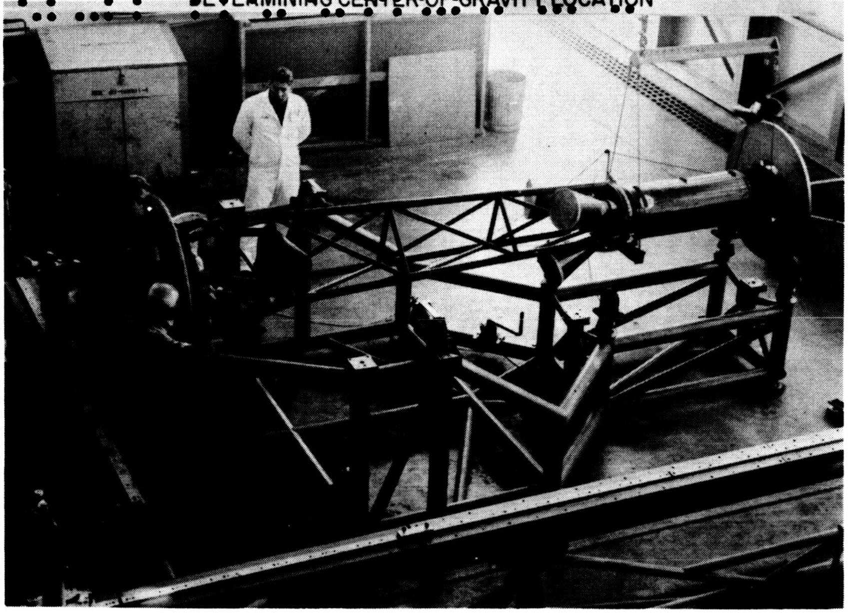
Figure 7

SPACECRAFT IN OPTICAL ALINEMENT FIXTURE FOR WEIGHING

LOD-61-3859

Figure 8

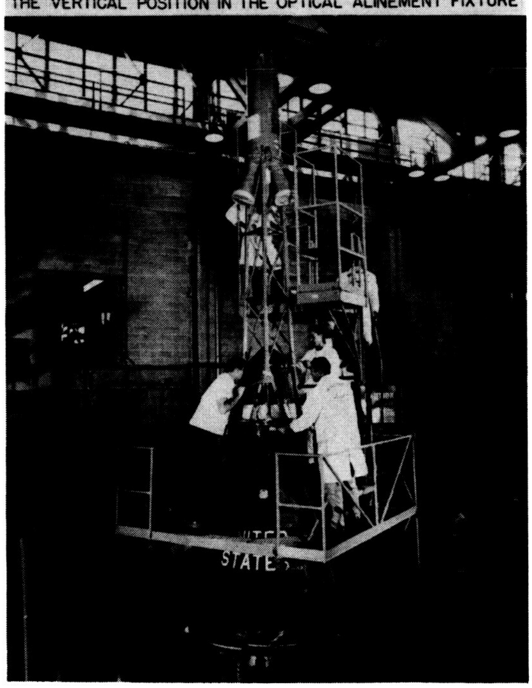
ESCAPE ROCKET IN THE BALANCE ALINEMENT FIXTURE FOR WEIGHING AND DETERMINING CENTER-OF-GRAVITY LOCATION



PL 60-59413

Figure 9

COMPLETELY ASSEMBLED SPACECRAFT AND ESCAPE TOWER IN THE VERTICAL POSITION IN THE OPTICAL ALINEMENT FIXTURE



LOD-61-3849

Figure 10

MERCURY-ATLAS COMPLEX OPERATIONS

DAYS TO LAUNCH											
11	10	9	8	7	6	5	4	3	2	1	0
11	Umbilical and Complex Check										
	10	Mechanical Mate									
		9	Spacecraft Systems Test								
			8	Interface and Abort Checks							
				7	Flight Acceptance Composite Test - Part I						
					6	Flight Acceptance Composite Test - Part II					
						5	Special Interference Tests				
							4	Radio-Frequency Aborts			
								3	H_2O_2 Tests		
									2	Spacecraft Service	
										1	Launch Count - Part I
											0	Launch Count - Part II

Figure 11

MERCURY-REDSTONE COMPLEX OPERATIONS

DAYS TO LAUNCH												
12	11	10	9	8	7	6	5	4	3	2	1	0
12	Umbilical and Complex Check											
	11	Mechanical Mate										
		10	Electrical Mate and Overall Test 2									
			9	Spacecraft Systems Test and Overall Test 1								
				8	Radio-Frequency Compatability - Part I							
					7	Pilot Insertion Simulation (Use Part II of Launch Count)						
						6	Overall Tests 3, 4, and 5					
							5	Overall Test 6				
								4	Simulated Flight			
									3	H_2O_2 Test		
										2	Spacecraft Service	
											1	Launch Count - (Part I
												0	Launch Count - Part II

Figure 12

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MERCURY FLIGHT-CONTROL FACILITIES AND OPERATION

By Christopher C. Kraft, Jr., and C. Frederick Matthews

NASA Space Task Group

INTRODUCTION

This paper presents some of the concepts and philosophies used in the design of the Mercury tracking and communications network and, in particular, the Mercury Control Center. In contrast to many data gathering and communications systems suitable primarily for postflight analysis, the Project Mercury facilities were designed to provide monitoring and flight control of the space vehicle in real time during each phase of the mission. In addition to a description of the facilities, the methods of performing flight control throughout the various phases of the flight are presented and some remarks on training and simulation aids are included.

It is not implicit that the Mercury flight-control plan should be considered as a blueprint for Apollo; however, it is hoped that this presentation will foster appreciation of the need for a sound operational approach. It will be evident that the facilities required for the relatively simple earth orbital mission are complex and that, in order to minimize complexity and ensure operational adequacy, the ground monitoring and control concepts and facilities required for Apollo will require sound and intensive planning fully integrated with the space-vehicle concepts and design.

MERCURY NETWORK

In a previous paper, Charles W. Mathews and Gerald W. Brewer noted some of the concepts and requirements for the Mercury network. Some of these should be reviewed as a basis for this presentation. When the design of the network was begun several years ago, determination of the orbital elements of an earth satellite, to indicate whether successful orbital insertion had been achieved, was an extremely difficult task and very often took many days to accomplish. Such long delays were unacceptable to Project Mercury because of the requirement to know, in real time, the present position of the Mercury vehicle and the impact point if reentry were initiated at any point should it become necessary. Therefore, the following basic requirements were formulated for the Mercury network design.

It was desirable (a) to maintain continuous real-time contact throughout the powered phase of the flight and the initial period following orbital insertion by means of radar tracking, voice contact with the astronaut, and surveillance of the vehicle systems and astronaut by telemetry; (b) to obtain similar contact periodically during each orbit; (c) to provide continuous real-time tracking during the normal reentry phase; and (d) to maintain continuous real-time impact point prediction. For conditions (b), (c), and (d) real time includes teletype delay time.

On the basis of these fundamental requirements, and many other criteria, such as the desire to use equipment already available and, of course, limitations as a result of economic considerations, the various tracking sites were chosen.

MERCURY CONTROL CENTER

Let us now turn to the basis for design of the Mercury Control Center at Cape Canaveral, Fla., and, subsequently, the control center at Bermuda, and the flight control aspects of the remote sites. The Mercury Control Center is considered as a focal point for the entire operation, particularly once the vehicle has left the ground. Prior to lift-off, the checkouts of the launch vehicle and space vehicle are conducted by separate crews. The assimilation of information on all other phases of the operation is carried out in the Mercury Control Center. This information includes the readiness of the launch vehicle, the preparations and checkout of the astronaut and space vehicle, the countdown preparations and readiness of the network, the preparedness of the recovery forces, the weather conditions in the launch and primary recovery areas and along the entire ground track, and, finally, the readiness of the flight-control teams in the Cape Canaveral and Bermuda Control Centers and the remote sites around the world.

After lift-off, in order that proper flight-control decisions can be made, certain data must be available and presented in the proper form to those performing control of the flight. These data include telemetered information regarding the launch vehicle, the space vehicle, and the astronaut, and voice communication with the astronaut, trajectory information on the performance of the launch vehicle, and the final conditions at power cutoff with regard to orbital insertion, or the actions to be taken should early cutoff or an abort occur. In addition, provision for initiation of certain commands to the vehicle were required. The content of each of these areas was a major factor in the design of the Mercury Control Center and a detailed analysis was required in each case to establish the specific data to be displayed or command to be provided. Only then could the detail layout of the operations room, consoles, and support areas proceed. (See fig. 1(a).)

In certain cases, these data, such as some space-vehicle systems displays, were identical to information displayed to the astronaut and were considered as backup. In other cases, such as aeromedical data and information on trajectory and insertion, these data were not available to the astronaut. Initial plans called for event commands to back up all automatic launch and reentry events. However, the astronaut was already provided with manual overrides to these programed events and, in the interests of simplicity for better reliability, this dual backup provision was deleted except for command of abort, changing the timer initiating retrofire, and direct command of reentry.

Idealistically, it would be desirable to have all this information presented to one individual or even to some automatic decision-making machine. However, because of the many different analyses that must be made in the relatively short time of powered flight, it would be impossible for one man to survey all the necessary data properly. Because this decision-making requirement involves many different disciplines which must be recognized and carried out, the responsibilities were divided as follows: In the rear of the room are three desks occupied by the directors of the various command functions of the operation. The NASA Operations Director sits in the middle and is responsible for directing all operational aspects of the project and makes the overall decisions leading up to the launch of the vehicle. The commander for recovery operations sits at his left and the commander responsible for network support at his right. The remainder of the positions in the Control Center are primarily responsible for the flight control of the mission after lift-off has occurred.

The Flight Director (fig. 1(b)) sits in the center of the next row and has overall responsibity for control of the flight from lift-off to landing. His duties are to coordinate the efforts of the flight-control personnel within the Control Center and throughout the network. In addition, it is through him that the decision would be made to abort or terminate the mission during this period.

To his right is the Network Status Monitor (fig. 1(c)) who acts as the test conductor for the Mercury network facilities. He conducts the network countdown and it is through him that the various facilities report their capability to support the flight test.

Seated next to the Network Status Monitor is the Launch Vehicle Monitor. (See fig. 1(d).) Certain telemetered parameters regarding the performance of the launch vehicle are presented to him on a continuous trace recorder. These data deal with the quantities measured by an automatic abort-sensing system within the launch vehicle. Although no direct abort action would be taken on the basis of the ground displays of these data, it is important to know in real time that some booster system failure may be imminent in order that the Mercury Control Center may prepare to take action should an abort actually occur.

The two consoles to the left of the Flight Director were originally to be used by a Range Safety Officer (fig. 1(e)) and a Recovery Status Monitor (fig. 1(f)). However, experience has shown that both of these functions were not required as a part of the flight control. These consoles, however, have been profitably used by personnel furnishing assistance to the Flight Director and the Operations Director concerning flight-control procedures, countdown information, and providing overall cognizance of the communications with other sites and supporting agencies.

The next row of consoles are the positions which perform the detailed flight-control functions. The position on the far left is occupied by the Support Control Coordinator (fig. 1(g)) who coordinates the efforts of all of the technical support required for the operations room including telemetry, all forms of communications, and other data transmission equipment such as that required to support the trajectory displays.

Next to him is the Flight Surgeon (fig. 1(h)) who has overall responsibility for all the aeromedical aspects of the flight during both the prelaunch countdown as well as the actual flight. It was recognized that, in addition to the basic aeromedical data displays, such as electrocardiogram, respiration, and body temperature, certain environmental system parameters were required for direct correlation of the astronaut's condition with his environment.

The Environmental Systems Monitor (fig. 1(i)), for convenience, sits next to the Flight Surgeon and is responsible for observing all data associated with the space-vehicle environmental system.

The Capsule Communicator, whose position is filled by one of the astronauts, is seated at the next console. (See fig. 1(j).) He is responsible for all communications to and from the astronaut and for keeping both the astronaut and the Mercury Control Center informed of the progress of the flight. Although all the flight controllers can monitor the astronaut communications, all normal voice reporting is carried out by the Capsule Communicator. In emergency situations, the flight controllers can talk directly to the astronaut but such conditions are considered rare and this privilege would be executed only when absolutely necessary. This type of discipline is necessary to avoid confusion. For reliability, redundant modes of communications are available including UHF, HF, voice on the command carrier, astronaut keying of the telemetry carrier, and Capsule Communicator keying of the UHF and HF carriers. In addition to conducting communications, the Capsule Communicator has event-indication light displays for both normal and aborted missions. These displays include built-in timers to warn of early or late events. These indications are somewhat similar to those provided the astronaut. In addition, a multi-channel television display is provided for overall reference of launch area activity, lift-off, and initial powered flight. Similar event and television displays are provided to the Flight Director who has, in addition, abort command capability.

The Systems Monitor (fig. 1(k)) observes the performance of the other major space vehicle systems, including the control systems and the electrical power systems.

Two consoles to the right of the room support the Flight Dynamics Officer (fig. 1(l)) and the Retrofire Controller (fig. 1(m)). These two engineers monitor the various trajectory displays during all phases of the flight. Originally, it was thought that these two functions could be performed by one individual but detailed analysis indicated that the volume and complexity of the data made this impractical. Therefore, the responsibilities were divided as follows. The Flight Dynamics Officer is concerned with the launch-vehicle performance and the important orbital-insertion parameters. The duties of the Retrofire Controller are to monitor the impact predictions made by the computer and to determine the various times of retrofire which may be necessary depending upon the conditions which prevail at launch-vehicle cutoff and during orbit. Details of the displays presented at these two consoles will be discussed later in conjunction with the computing facilities required to support the flight.

In addition to display meters at the various consoles, continuous trace recorders are provided to the Flight Surgeon, Environmental Monitor, Systems Monitor, and Launch Vehicle Monitor to provide time histories of certain of the measured quantities.

The wall map at the front of the room presents the ground track of the flight on which the real-time position of the space vehicle is plotted. Also, the position of the various remote sites is given with the overall status of each site and its equipment presented in the form of colored circles and symbols on this map. The map furnishes quick overall status of the network and the flight to the controllers and the Operations Director. Subsequent to monitoring the launch in real time by means of telemetry and voice reports, each flight controller continuously monitors information obtained from the remote sites as the astronaut traverses the orbit and makes recommendations to the Flight Director on the basis of these data. To aid in this assessment, the areas adjacent to each side of the map are used to plot the pertinent data obtained from each of these remote sites so that trends of various measured quantities can be determined.

BERMUDA SITE

The site at Bermuda was designed and is operated with the same concepts as those used at Cape Canaveral in that it was considered to be a backup to the Mercury Control Center in certain instances such as command control following an abort decision. Figure 2 shows an overall view of the operations room at the Bermuda site. It can be seen that the layout and the consoles are similar to those at the Mercury Control Center at Cape Canaveral, Fla. No further details will be presented here except to note that certain additional computing facilities are provided at the Bermuda site because of the necessity to perform the backup function of determining the conditions of the orbit or certain times of retrofire associated with aborts.

REMOTE SITES

The remote sites were primarily designed to provide information to the Mercury Control Center on the status of the astronaut and the space vehicle, to keep the astronaut updated on mission progress, and to provide tracking information to the computer. A picture of a remote-site operations room is shown in figure 3. There are three flight controllers at each site supported by approximately 20 to 30 engineers and technicians, the number being dependent upon the detailed equipment at the particular site. Each site has a Capsule Communicator, an Aeromedical Monitor, and a Systems Monitor. The duties and displays of these monitors are essentially equivalent to their counterparts in the Mercury Control Center with the exceptions that the function of the Environmental and Systems Monitors are combined, and at command sites the command controls are on the Capsule Communicator's console.

COMPUTING AND DATA FLOW

Now the facilities and data required to perform the flight-control function will be considered. Perhaps the most important and by far the most complex system is the computing and associated data flow facilities required.

Figure 4 presents the various sources of tracking data and computing systems which are required during launch, orbit, and reentry to provide the necessary information on trajectory, time of retrofire, and impact prediction. All the computations necessary for the flight are performed by computers at the Goddard Space Flight Center near Washington, D.C., by using two IBM 7090 electronic data processing systems for redundancy. The method in which data are supplied to these computers is dependent on the flight phase. Because of the critical nature of the launch and orbital insertion period, multiple sources of data are provided to ensure decision and action capability should a data failure or aborted flight occur. The Atlas launch vehicle is tracked and controlled by the GE-Burroughs guidance system. This source of data is the most accurate of those available and therefore is considered primary for position and velocity data which are transmitted by means of high-speed data lines to the computers at the Goddard Space Flight Center. In addition, other data such as velocity, time to go, and certain other guidance parameters are supplied directly to the Mercury Control Center. The Azusa system also tracks the launch vehicle and is used as a backup to GE-Burroughs data.

Once the launch vehicle cuts off, either in the abort or normal case, it is desired to track the space vehicle rather than the launch vehicle; therefore the source is switched to FPS 16 data either through the IBM 7090 impact predictor, or direct. The GE-Burroughs data can also be supplied directly to the plot boards but retrofire time would be lost as this time is determined only by the computers at the Goddard Space Flight Center. In addition to the tracking data, event information occurring during the launch is provided to the Goddard computers from the Mercury Control Center.

The data from the Bermuda site from both FPS-16 and Verlort radars provide similar trajectory information to the Bermuda Flight Controllers through an IBM 709 electronic data processing system. In addition, the data from this computer, or directly from the radars, are supplied through an automatic teletype transmission system to the Goddard Space Flight Center.

Once the spacecraft is in orbit, data from the remote-site radar systems are transmitted by teletype to the Goddard Space Flight Center and automatically processed in the computer to update the various displays at the Mercury Control Center at Cape Canaveral, Fla. Also, once the reentry maneuver has taken place, this tracking system provides data for accurate impact prediction.

PLOT-BOARD DISPLAYS

The data obtained from these facilities are presented in the Mercury Control Center to the Flight Dynamics Officer and Retrofire Controller in both plot-board and digital display form. Figure 5 is a photograph of the consoles of the Retrofire and Flight Dynamics Officers and includes the four plot boards used for displaying trajectory and impact prediction information during launch insertion, orbit, and reentry. As shown in figure 6, plot board no. IV on the far left is used to present impact prediction. Plot board no. II presents altitude and cross-range deviation as a function of down-range distance.

Plotted on plot board no. III (fig. 7) are velocity and acceleration as a function of elapsed time. After a certain elapsed time, plot board no. III switches to plotting the yaw-error signal to the launch vehicle and the predicted insertion altitude, that is, the altitude to which the computer is guiding the launch vehicle as a function of time to go to cutoff. The most important chart is that of plot board no. I which plots flight-path angle against velocity ratio, that is, the ratio of the present velocity to the velocity required for orbit at the desired altitude. This plot is presented to the Flight Dynamics Officer in three different scales; when the velocity ratio reaches 0.9, the greatly magnified third scale allows the important go—no-go decision to be made at the time of launch-vehicle power cutoff.

During powered flight, the Flight Dynamics Officer uses these charts to determine whether the launch vehicle is performing satisfactorily and, in addition, uses these data to determine the time for abort should an abort for astronaut or space vehicle systems malfunctions become necessary. During the last 15 seconds of powered flight, he concentrates on the magnified plot of flight-path angle. From this plot he can determine that the proper orbital parameters have been achieved and at the same time the computer presents its go—no-go recommendation by means of lights on this same plot board. Also, at the same time, the computer presents the times of retrofire associated with the cutoff conditions achieved to the Retrofire Controller; that is, if an abort or no-go decision is made, the time of retrofire required to land in a particular recovery area will be displayed, and if a go condition is reached, the time of retrofire at the end of a normal three-orbit mission and the time at the end of each intermediate orbit will be displayed.

Subsequent to power cutoff, the plot boards continue to give data in various forms on the position of the space vehicle and certain other parameters related to reentry from orbit. The particular plots displayed depend upon the decision made at cutoff and they are not included in the simplified diagrams in this paper. As data are received from the remote sites, the times for retrofire and the other trajectory information are continually updated. In addition, the computer supplies to each site, by means of teletype, acquisition information and updated retrofire times for normal and contingency recovery areas.

After retrofire, details of the event are transmitted from the remote sites by teletype and manually inserted into the computer to provide more accurate impact-point prediction.

COMMUNICATIONS

As can be seen from this discussion of information and data flow, a highly complex communications system is required to transmit radar data and information messages to and from sites around the world. The entire communications network to accomplish this task is a subject of its own and cannot be discussed here in detail; however, certain aspects of the system are necessary to the understanding of flight control. Teletype is provided to all sites and uses redundant paths for high reliability and multiple paths for high density traffic during certain phases. Teletype has an inherent lag, however, and requires an average of 6 to 10 minutes to transmit a message to any given site and receive a reply. Voice communications are provided to all command sites, that is, Bermuda; Muchea, Australia; Hawaii; Guaymas, Mexico; and the California site and, in addition, Woomera, Australia; and all other sites in the continental United States. Experience to date has shown that fairly reliable communications can be maintained although at certain times there are serious propagation problems. Voice communication with all sites would be very helpful and would provide for much simpler operational procedures but it is not considered a necessity.

PLANNING OF SPACECRAFT-NETWORK OPERATIONS

There are many aspects of the Mercury network such as selection and construction of sites, instrumentation tests, countdowns, and so forth, which should be discussed but, because of the limited scope of this paper, cannot be included. However, it is emphasized that the network and the operational planning required should be considered in the design of any space vehicle. If this is not done, the ground support necessary to the mission will be less than desired and, in addition, its completion will lag the space vehicle and, therefore, tend to pace the program.

In order to specifiy adequately the operational support required, a clear definition of the respective and interrelated roles of the ground network and the space vehicle and its occupants must be provided to support all development and operational missions and test objectives.

Once these roles are clearly established, the onboard measurements and instrumentation required to support the role of the ground flight control can be defined simultaneously and in conjunction with the definition of the space-vehicle crew displays and controls. In accomplishing this result, there should be a clear-cut delineation of these operational flight control measurements and instrumentation as distinct from the often varying and sometimes "ad hoc" instrumentation used solely for postflight analysis. In addition, since this operational instrumentation is used in real time, it must be compatible with real-time ground displays at various sites and this condition suggests that the same standards of instrumentation specifications as used in the cockpit displays are required in terms of replacement interchangeability and calibration.

FLIGHT CONTROL

In an attempt to give a better understanding of how the control of the flight is achieved the rest of the paper will deal with the details of how flight control is performed during each phase of the flight. Of course, it must be recognized that an orbital flight on which these procedures will be exercised has not been accomplished as yet and the information to be presented is the result of a number of simulated flight tests in which the astronauts and the flight controllers have participated. However, the Redstone launches and the Atlas suborbital flights have been used to demonstrate these techniques and, with minor changes, appear to be adequate. In fact, the Redstone flights were planned on the basis of an orbital launch in order to exercise these techniques.

Figure 8 is a pictorial presentation of the important launch phase and orbital insertion. Just prior to lift-off, final confidence checks are made with the astronaut to confirm communications and proper systems functioning. The final phases of the countdown and the lift-off are transmitted to the astronaut by the Capsule Communicator in the Mercury Control Center. At lift-off, the astronaut confirms that the onboard clock and timing devices have started and that communications are still satisfactory. During this early phase of the flight, trajectory information from range safety plot boards is transmitted by voice to the Control Center to assure that the proper trajectory is being followed. The astronaut makes a communications report every 30 seconds and indicates such quantities as acceleration, oxygen, and cabin pressure. The ground flight controllers continuously monitor the various displayed parameters during this period.

One of the most critical points in the launch occurs at about 30,000 feet or at about 70 seconds after launch at which time the cabin pressure seals and is maintained at approximately 5 pounds per square inch. A communications procedure with the astronaut is used to ensure that, if the cabin pressure and suit pressure were to fail, abort action by the astronaut or the Flight Director would be taken to prevent an excessive altitude being reached by the spacecraft. Continued surveillance is made by the monitors, and the astronaut is kept informed of the trajectory and the status of the flight from both a launch vehicle and spacecraft point of view. Staging of the Atlas is confirmed by both the astronaut and the ground controllers and 20 seconds later confirmation is given by the astronaut that the escape tower has jettisoned properly. In all cases of programed spacecraft events, the astronaut is prepared to perform manual backup should it be required. After this period while the launch vehicle is being propelled by the sustainer engine, all the flight controllers are making a close analysis of space-vehicle systems performance and astronaut conditions so that, at about 4 minutes and 30 seconds, a final go—no-go decision can be made in conjunction with the astronaut. This decision is a commitment to orbit of the astronaut and spacecraft systems. Certain ground rules upon which this go—no-go decision is made are formulated and agreed upon many weeks in advance of the flight.

From this point on, almost complete attention is given to launch-vehicle performance in the form of trajectory displays to ascertain that the proper values of orbital parameters are achieved, that is, the proper velocity, flight-path angle, and altitude. At insertion into orbit, the conditions which are achieved are immediately transmitted to the astronaut and the astronaut confirms that proper separation of the spacecraft has been achieved and that a turnaround maneuver has been properly initiated and maintained. Should a go decision be achieved the astronaut would be informed either by the Mercury Control Center or the Bermuda site of the time of retrofire calculated by the computer for a normal reentry at the end of the design mission. If the computer indicates a no-go condition, the astronaut would be so informed and the time of retrofire necessary for landing in a preferred recovery area would be indicated to him and to the Bermuda flight controllers. The actions to be taken by ground control and the astronaut would, of course, be dependent upon the final cutoff conditions. These actions can be extremely time-critical and the communications procedures to be followed have been worked out by intensive review and training in this particular area.

The insertion takes place about midway between Cape Canaveral and Bermuda and, approximately 1 minute after this time, control of the voice link and command to the spacecraft is transferred to the Bermuda site. The Bermuda site actually acquires the spacecraft radio frequency links at about 1 minute and 30 seconds before cutoff.

Summarized pertinent data regarding the launch phase are immediately transmitted by voice and teletype to all remote sites from the Mercury Control Center. The Recovery Control Center, located adjacent to the flight-control operations room, monitors all of this information and they are kept informed in almost real time of the expected landing points of the spacecraft.

Let us now consider that orbit has been achieved and examine the activities at a typical remote site. Figure 9 outlines the procedures followed during a normal orbital pass which, if passing directly overhead, will last up to approximately 6 minutes. Prior to contact, the station receives messages on astronaut and systems performance from the Mercury Control Center and other sites, and acquisition messages and certain times for retrofire from the computer. The acquisition messages are nominally received 45 minutes, 25 minutes, and 5 minutes prior to the horizon time of the spacecraft. Upon making the initial voice contact, the astronaut immediately reports the overall status of himself and the spacecraft. If all is well, he is then given the updated times for retrofire for the end of this particular orbit and the intermediate contingency recovery areas in order that he may have the most up-to-date information should an emergency develop at any time.

Because of the importance of the accuracy of the spacecraft clock, the astronaut indicates his elapsed time since lift-off and the time of retrofire set in the clock. These times are compared with the telemetered values and transmitted back to the computer at the Goddard Space Flight Center and used in future calculations of times for retrofire. During this time, radar data from the site are being transmitted to the same computer for updating the orbit characteristics. The astronaut is informed of the progress of the mission and is kept updated on certain matters concerning recovery and worldwide weather. Then, if necessary, the astronaut makes a detailed status report to the site regarding systems functions such as any changes he may have made in mode of operation, his physical condition, any communications phenomena or problems, how he has been able to navigate, and anything concerning mission control which should be reviewed. Also, over some sites certain tests will be made in conjunction with the ground to evaluate

[redacted]

further the astronaut's ability to perform in a space environment. After loss of contact with the spacecraft, the site immediately transmits a summary message to the Mercury Control Center and all other sites regarding the status of the astronaut and spacecraft systems.

This process is followed from station to station as the spacecraft continues to orbit. Of course, certain other procedures with regard to clock setting and retrofiring are followed at command sites, such as Muchea in Australia, and the California site. The normal retrofire maneuver is performed by a countdown from the ground to the astronaut and retrofire would be initiated simultaneously by the astronaut and the Capsule Communicator. The onboard retrofire timer will be used primarily as a backup to these commands. The mode of attitude control of the spacecraft throughout orbit, retrofiring, and reentry will vary from flight to flight in order to evaluate both the astronaut's capability and the systems performance.

TRAINING AND SIMULATION

In order to train both the astronaut and the ground crews in flight-control problems and to develop the procedures necessary for flight control, facilities were constructed both at the Mercury Control Center and at the Langley Research Center which allow complete flight simulation. At both of these facilities a procedures trainer is provided in which not only the normal mission can be flown but almost any conceivable malfunction can be simulated. Outputs from these trainers are provided to the flight-control consoles and complete realism can be obtained by introducing such problems as space vehicle and astronaut malfunctions, telemetry and radio noise, signal dropouts, failed readings, and so forth. In addition, at the Control Center, complete trajectory simulation is combined with this operation so that the entire launch orbit and reentry phases can be duplicated. These facilities have proven to be extremely useful to the operation and, as has been stated by the astronauts, are considered to be one of the most important training devices to be developed for Project Mercury. Such facilities will be a mandatory requirement for the Apollo program. These facilities should be considered during the design phase of the actual vehicle so that ground crew and astronaut training can begin as soon as possible.

CONCLUDING REMARKS

It is realized that only a cursory look has been given to the Mercury network and the flight-control aspects of the project.

Engineers from the divisions responsible for these aspects of the project are devoting time to the Apollo program and the Apollo contractor is urged to take advantage of the experience of these people with the Mercury Project.

[redacted]

DECLASSIFIED

MERCURY CONTROL CENTER

Figure 1(a) S-61-59

FLIGHT DIRECTOR

Figure 1(b) S-61-60

NETWORK STATUS MONITOR

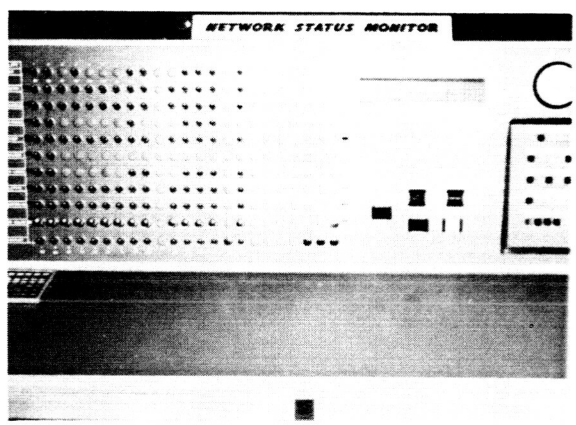


Figure 1(c) S-61-61

[redacted]

MISSILE TELEMETRY MONITOR

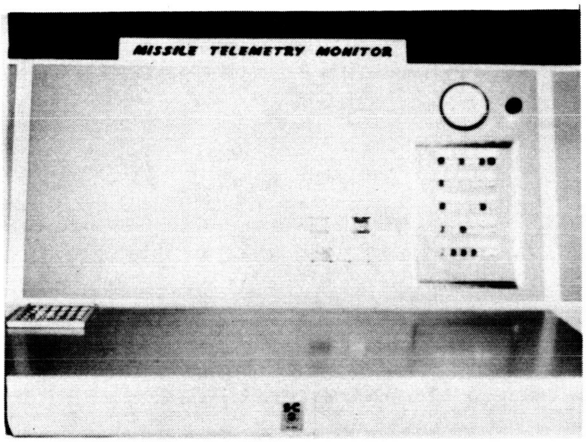


Figure 1(d) S-61-62

PROCEDURES

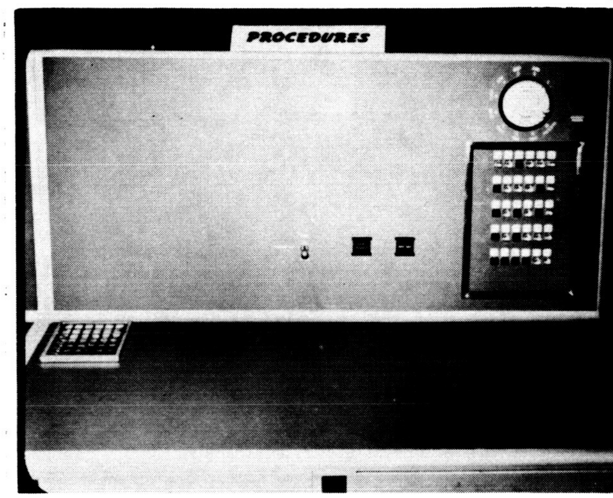


Figure 1(e) S-61-63

PROCEDURES

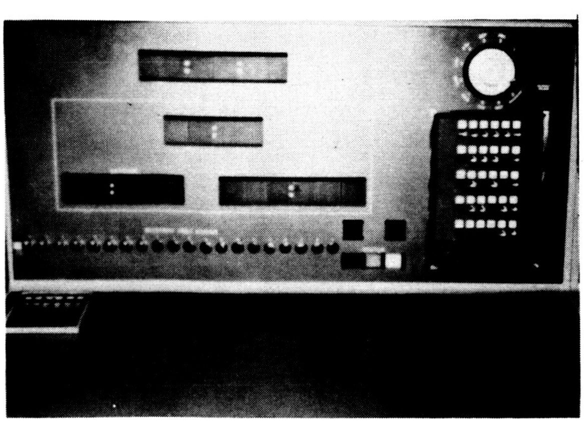


Figure 1(f) S-61-64

SUPPORT CONTROL COORDINATOR

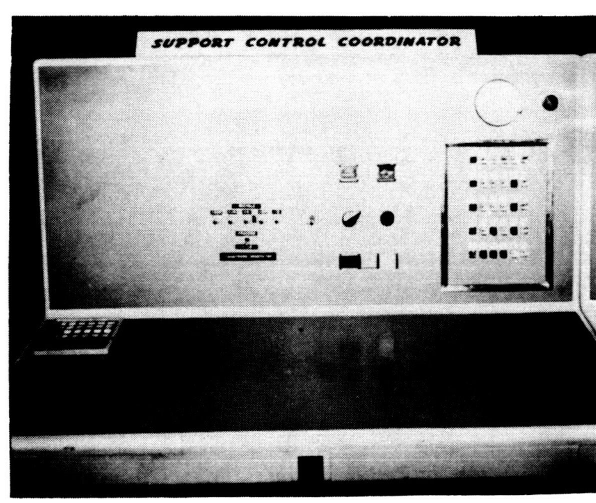


Figure 1(g) S-61-65

[redacted]

FLIGHT SURGEON

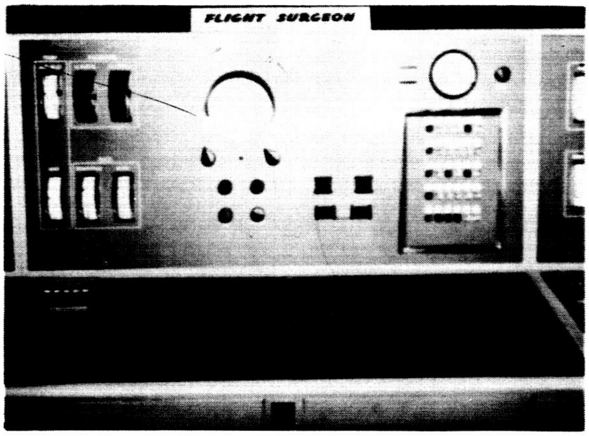


Figure 1(h) S-61-66

CAPSULE ENVIRONMENT MONITOR

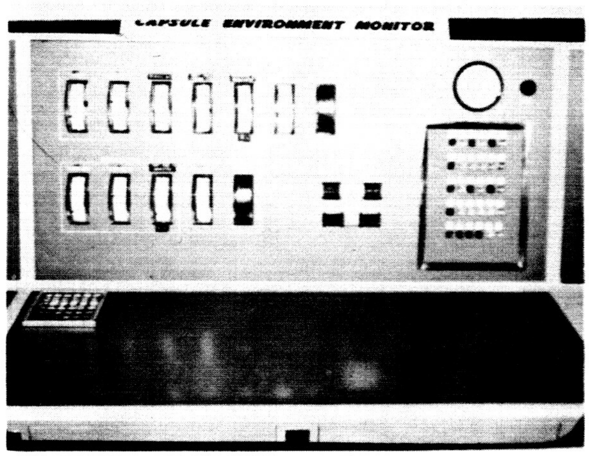


Figure 1(i) S-61-67

CAPSULE COMMUNICATOR

Figure 1(j) S-61-68

CAPSULE SYSTEMS MONITOR

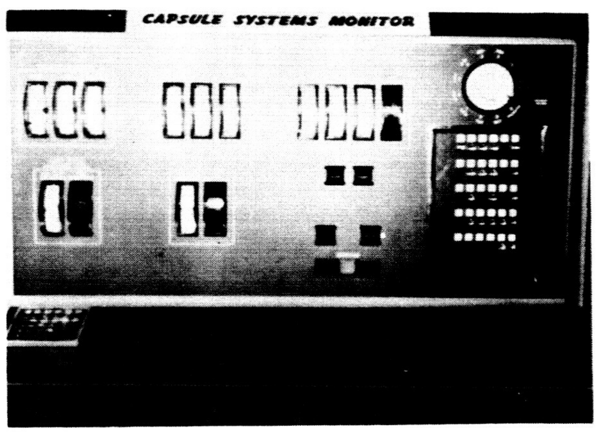


Figure 1(k) S-61-69

[redacted]

FLIGHT DYNAMICS OFFICER

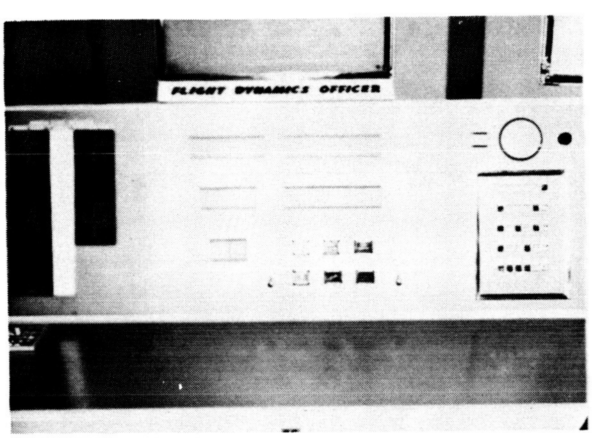


Figure 1(l) S-61-70

RETROFIRE CONTROLLER

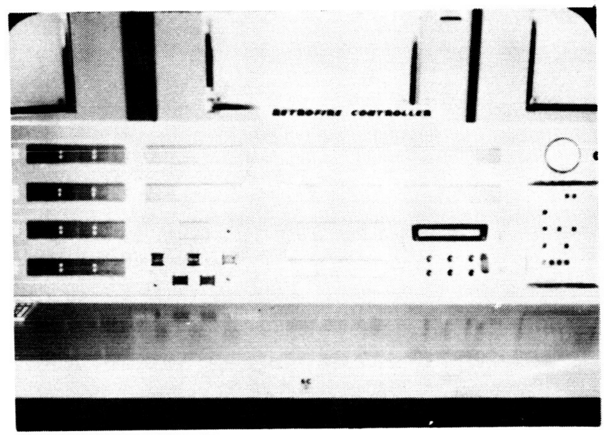


Figure 1(m) S-61-71

BERMUDA OPERATIONS ROOM

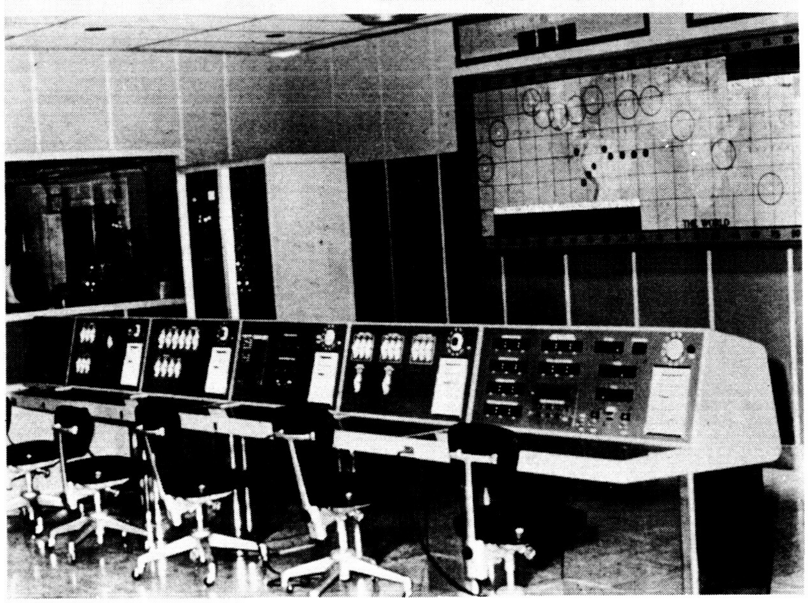


Figure 2 S-61-42

[redacted]

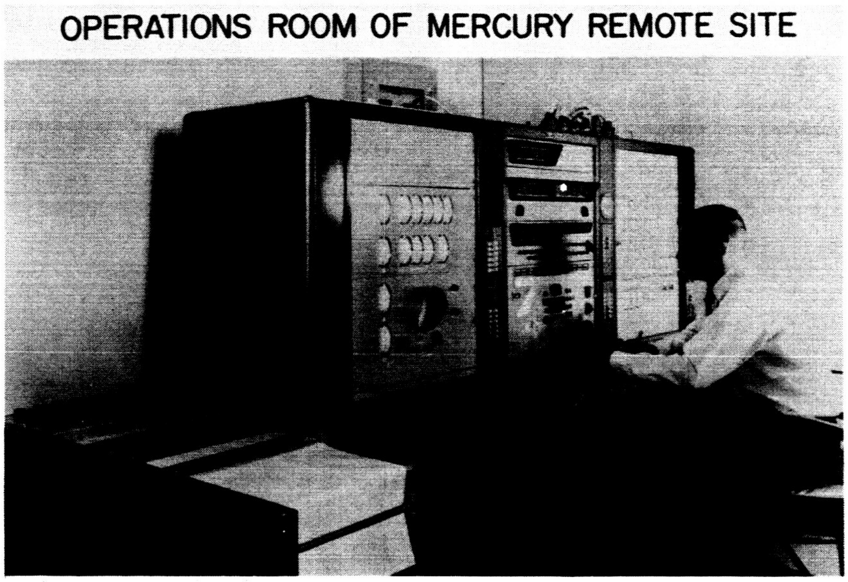


Figure 3 G-383

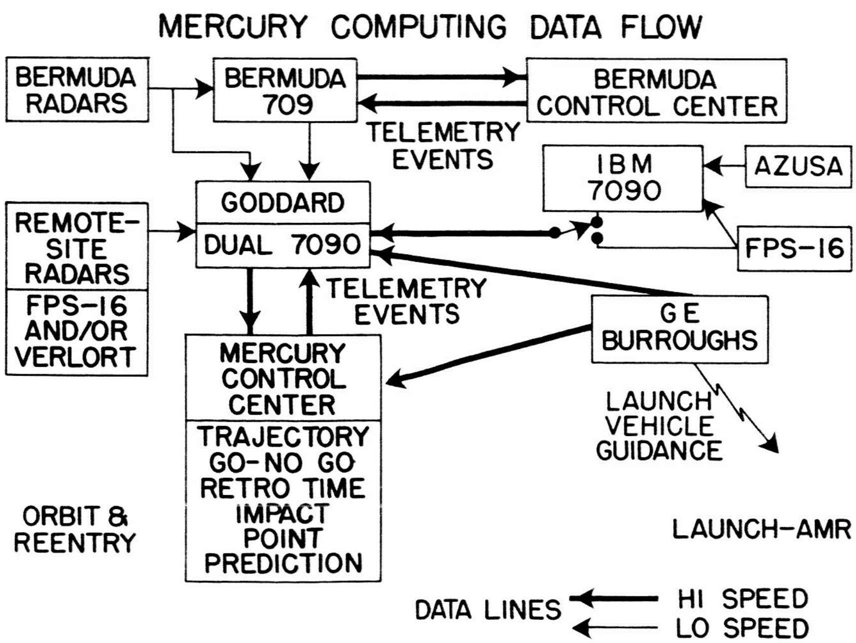


Figure 4

CONFIDENTIAL

RETROFIRE CONTROLLER AND FLIGHT DYNAMICS OFFICER CONSOLES AND PLOT BOARDS

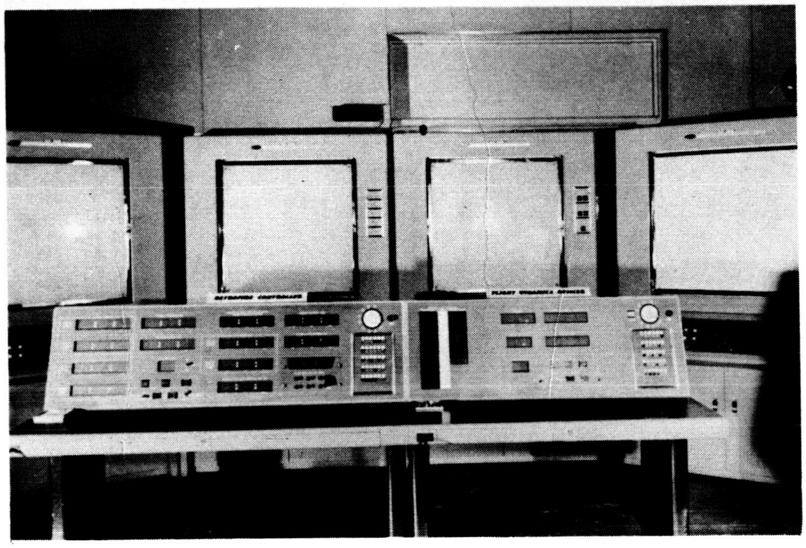


Figure 5 S-61-41

MERCURY CONTROL CENTER PLOT-BOARD DISPLAYS

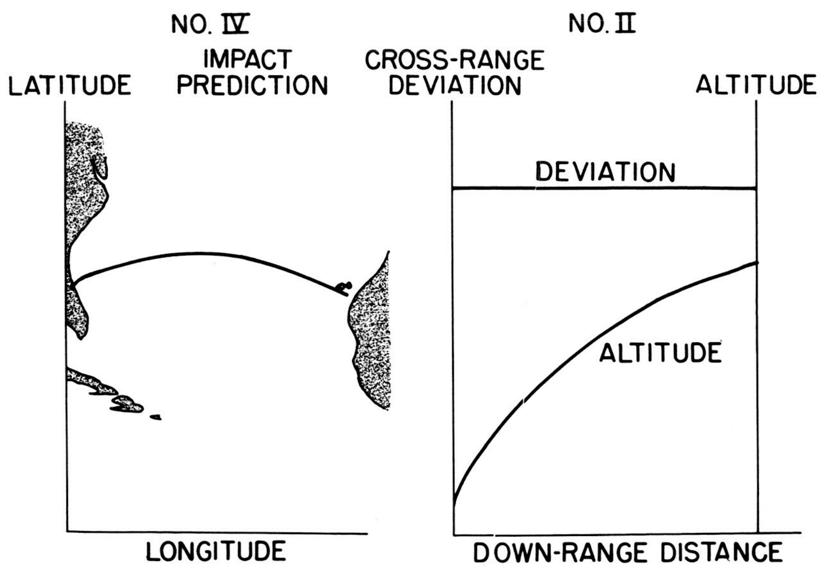


Figure 6

[redacted]

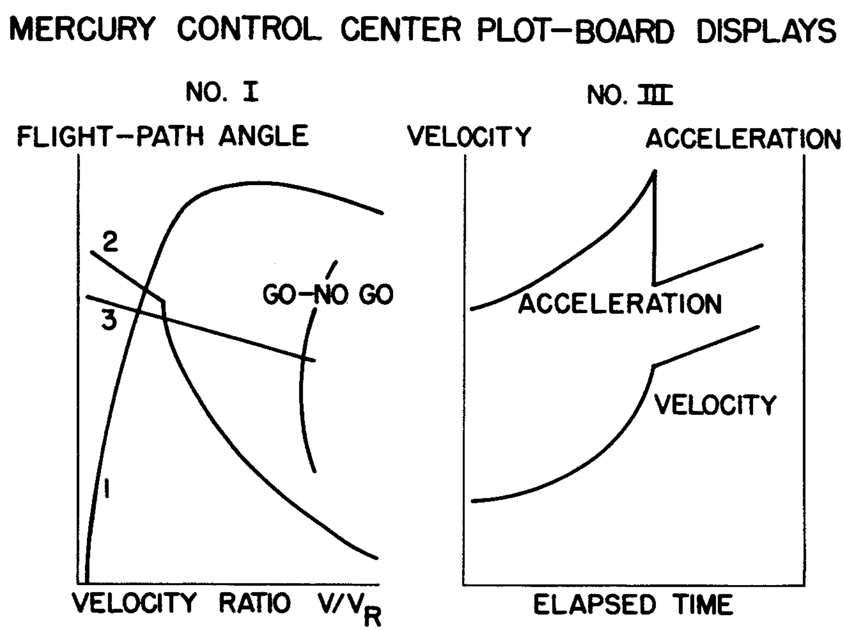


Figure 7

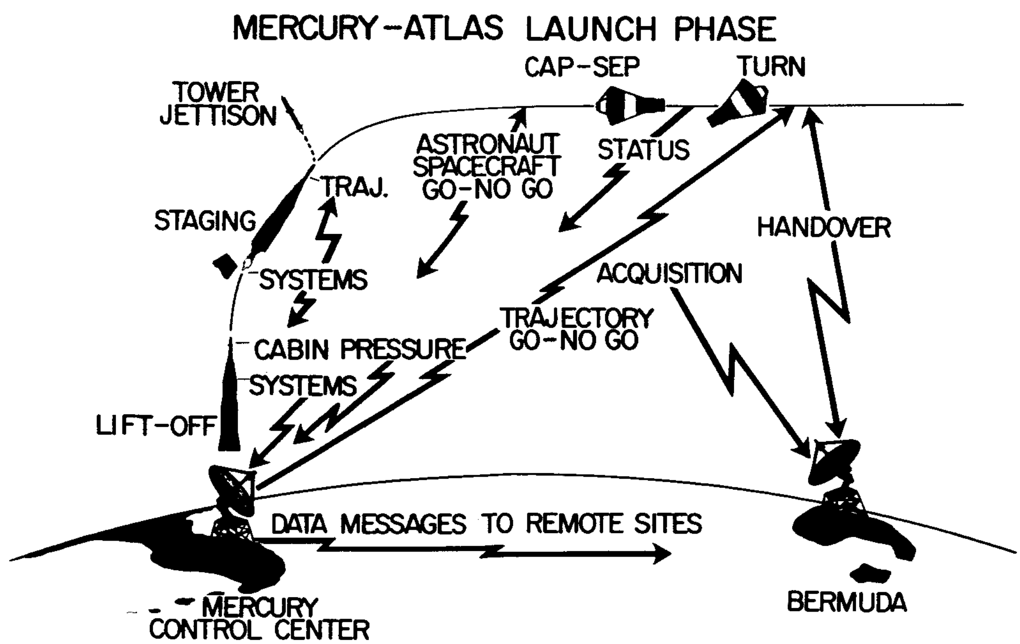


Figure 8

NORMAL PASS OVER TYPICAL REMOTE SITE

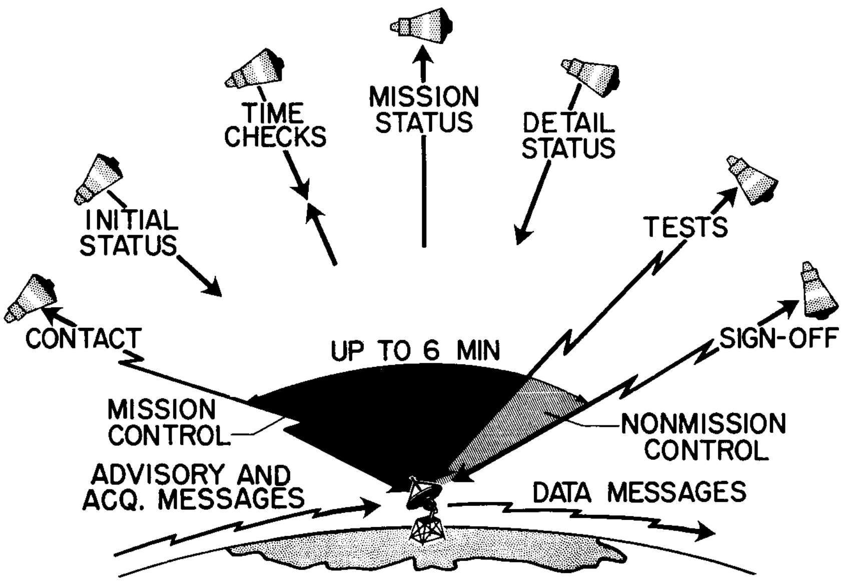


Figure 9

[redacted]

MERCURY RECOVERY OPERATIONS

By Robert F. Thompson, William C. Hayes, Jr.,
and Donald C. Cheatham

NASA Space Task Group

INTRODUCTION

Recovery operations are generally defined as the support required for location and retrieval of the astronaut and spacecraft at the termination of a mission. This support must consider both normal flights and possible aborts; however, it should be noted that for Project Mercury, recovery of abort modes having reasonable probabilities of occurring imposes by far the greatest requirements on recovery forces. Although it is recognized that, as flight-vehicle reliability increases, these recovery support requirements can be decreased; it is also reasonable to expect that vehicle systems in the near future will require rather extensive abort recovery support for manned operations. This paper presents a description of the general scope of recovery support as currently planned for Mercury orbital flight, a description of recovery vehicles and techniques to be used, and a brief review of experiences gained during suborbital flights to date. 44

It should be noted that the recovery forces which support Project Mercury, the airplanes, ships, helicopters, and special launch-site vehicles are provided by the Department of Defense and in most cases are the routine operational units that can devote only a relatively small part of their time to Mercury recovery. Therefore, the special techniques and equipment developed for recovery have been purposely simplified insofar as practicable in order to minimize the training and logistical efforts required of these forces. This, coupled with the relatively low frequency of flight operations to date, has permitted the recovery forces to support this program with a minimum of diversion from their normal defense functions. This point must be considered in any planned use of defense forces for recovery in future operations.

Planning for manned orbital space-flight operations as conceived for Project Mercury, wherein general consideration was given to the expected reliability of the over-all flight-vehicle system, has indicated the need for (1) a rapid crew egress and pad-rescue capability during the late countdown and early powered phase of flight, (2) a positive short-time recovery capability throughout all phases of powered flight and for landing at the end of each orbit, and (3) a considerably reduced recovery deployment in support of unplanned landings along the orbital track.

GENERAL RECOVERY CATEGORIES

Recovery areas are considered in two broad categories: first, North Atlantic areas which require recovery support if it becomes necessary to abort the mission at any time during launch, shown as areas A to E in figure 1, or to land at the end of any one of the three orbits currently planned, areas F, G, and H in figure 1; and, second, contingency recovery areas which basically lie along the remainder of the ground track outside the North Atlantic areas and which would contain any emergency landings from orbital flight.

In order to minimize as far as possible the recovery areas associated with aborts on or near the launch site, winds are measured just before lift-off and these measurements are put into a programed computer to determine a specific landing corridor from the launch pad out to deep water. The launch-site forces are then positioned to support recovery in this area.

For the drag-type vehicle under consideration, programed use of the retrorockets at the higher launch speeds allows some reduction in over-all abort launch areas. The aiming for specific landing points for aborts at higher launch speeds combined with high probability dispersion about these points determine the size and distribution of the abort-launch landing areas. For current Mercury planning, if flight vehicle and network systems function basically as planned for an aborted flight, about 95 percent of all landings in area E would be contained in an ellipse having a major axis of about 255 nautical miles and a minor axis of about 22 nautical miles. For landing at the end of any orbit, the high probability landing areas would be contained in ellipses having major axes of about 150 nautical miles and minor axes of about 45 nautical miles.

In the North Atlantic areas, recovery forces will be predeployed to provide a short-time location and retrieval capability. By "short time" it is meant that the areas defined by high probability dispersion will contain forces so distributed as to have the capability of recovery with specified times. Currently it is planned to provide those landing areas having a higher over-all probability of use with a maximum recovery time of 3 hours and some of the lower probability landing areas with a maximum recovery time of approximately 6 hours. Unmanned orbital flights are supported by forces in these same areas; however, access times are increased to about 9 hours in order to minimize recovery-force requirements.

For contingency recovery, it is desirable to search for the spacecraft during the time period that onboard location aids will remain active by using specially equipped search airplanes which could be airborne shortly after an emergency landing is reported. Therefore, recovery support in these areas will be provided by airplanes strategically deployed to provide efficient location. Retrieval support for this phase will utilize facilities activated after the fact.

LOCATION TECHNIQUES

In the launch-site area where a land or shallow-water landing due to an escape tower abort before or shortly after lift-off occurs, location is provided from visual contact by the recovery forces. The Local Recovery Force Commander is airborne in a helicopter behind the launch pad at the time of launch and other launch-site support forces are prebriefed and deployed. If an abort is initiated, it will be observed by these forces who will then proceed to carry out the retrieval assignments.

In the event of an unobserved landing, it is desirable that the spacecraft present an active target regardless of its position in the water or on the ground. Therefore, the spacecraft has been equipped with the proven location aids shown in table I which have an adequate operational range and which are commercially available. Although search aircraft must be equipped with special receivers, the minor logistic problems have been compensated by the increased search capability and reliability of the aircraft.

In the deep-water areas of the North Atlantic, search airplanes will be airborne in the recovery areas at the time of launch. If required, the airplanes can be directed to within range of the spacecraft ultra-high-frequency/direction-finder (UHF/DF) beacons by landing prediction information provided by the tracking-computing network. Also useful in this general landing-area category are: SOFAR, the sound fixing and ranging system which utilizes an underwater-detonation technique; high-frequency/direction finder (HF/DF), which is activated about 2.5 minutes after landing; and radar chaff, which is jettisoned at the time of drogue parachute deployment. For various operational reasons, these latter systems are considered secondary; however, they could be of value in the event of some failure of the network to provide landing-area information.

When within UHF/DF range, the search airplanes home on the spacecraft electronic aids until establishing visual contact by using such aids as the fluorescein sea marker, the spacecraft structure, or the flashing light at night.

To provide spacecraft location in contingency recovery areas, airplanes will be on strip-alert at strategic positions, and techniques similar to those established for landings in the deep-water areas of the North Atlantic will be followed. Contingency recovery planning has utilized a location time parameter of 18 hours in determining numbers of aircraft to be deployed and lifetime requirements for spacecraft electronic aids. In addition, the search airplanes are equipped with buoys containing UHF/DF beacons which may be dropped in the vicinity of the spacecraft in order to maintain contact after the spacecraft location aids have been depleted.

RETRIEVAL TECHNIQUES

Once the landing site has been identified, the next phase of recovery is retrieval. A North Atlantic retrieval will be accomplished by helicopters which are based aboard ship in the planned landing areas, ships, or special launch-site vehicles.

For contingency recovery areas, the capability of rendering emergency on-scene assistance has been provided; however, as previously noted, retrieval would be accomplished by utilizing whatever forces could be made available after a contingency recovery requirement developed.

It is important to note that for Project Mercury the retrieval vehicles in the North Atlantic areas all have the capability of transporting the entire spacecraft. This capability has been very helpful in planning for the improbable event that the astronaut would be incapacitated and require recovery while still in the spacecraft. The capability of retrieval vehicles to lift the entire spacecraft would be desirable in any future programs; however, this will of course become increasingly difficult as the size and weight of the spacecraft increases. Therefore, such factors as sea-keeping ability, fatigue life, ease of access to crew, or perhaps even a modular design to facilitate retrieval should be considered.

In general, retrieval can be accomplished by any of the following vehicles through any one of several modes of operation; however, these vehicles will be discussed on the basis of the standing operating procedures developed for Project Mercury.

Helicopter

Two phases of retrieval by helicopter are shown in the photographs in figures 2 and 3. The retrieval technique is as follows: As the helicopter hovers above the spacecraft a crewman cuts the 16-foot HF antenna with specially designed cutters and engages a lifting loop located on top of the spacecraft with a hook. Hook engagement is facilitated by a long, detachable pole and the lifting hook has been prerigged through a lifting cable to the helicopter external cargo sling. When engaged, the attaching handle is withdrawn from the hook and the helicopter is free to lift the spacecraft. For manned flights, if the astronaut so desires, he may transfer from the spacecraft to the helicopter using the "horse-collar" and rescue winch technique before the spacecraft is lifted. If for any reason the astronaut does not desire to transfer at this time, he may remain in the spacecraft and be transported back to the recovery ship.

This technique was chosen from several tried as having the advantages of simplicity and over-all reliability. Note that a positive manual attachment is possible without placing a helicopter crewman in the water.

Ships

A typical retrieval operation by ship is shown in figures 4 and 5. The ships of main interest in the recovery phase of the Mercury program have been destroyers because these are the types of ships which are available in sufficient numbers to provide the magnitude of support required. Fortunately they also have other very attractive features such as high-speed and good communication capabilities. Here again the lifting line is attached to the spacecraft by the long-handled lifting hook and then fair-led through a boat davit to an existing deck winch. For rough-weather retrieval a hold-off arm is added to the davit to prevent contact between the ship and spacecraft. The spacecraft is lifted clear of the water as shown in figure 5, swung inboard, and placed on the deck beneath the davit.

Lighter Amphibious Resupply Cargo (LARC)

This amphibious vehicle, shown in figure 6, which along with helicopters, makes up the primary retrieval support in the launch-site area, was chosen from the U.S. Army inventory and has the capability of negotiating the sand, brush, swamp, shallow water, and surf characteristic of the area with comparative ease. For Mercury, a hydraulic hoist capable of lifting the spacecraft and equipment for fighting local fires which could result from spacecraft onboard systems have been added.

Launch-Pad Rescue Vehicles

The full-tracked, armored personnel carrier, shown in figure 7, is to provide launch-pad rescue support in the event of a spacecraft landing in the vicinity of a live or burning launch vehicle. This personnel carrier is equipped to push the spacecraft clear of the launch vehicle and to give protection for the pad-rescue team in the event of explosions or fires.

Another facet of pad rescue is retrieval of the astronaut from the spacecraft in the event of a mission postponement where it is not desirable to activate the escape tower system and where the gantry and service towers have been withdrawn. For this type of pad rescue the cherry picker, shown in figure 8, has been provided and allows the pad-rescue team to go rapidly up to the spacecraft and assist in astronaut egress. It also has a remote control capability, so that the astronaut can egress from the spacecraft to the cab and descend without assistance.

In the event of an emergency landing in the water in a contingency recovery area, an auxiliary flotation device (fig. 9), which may be dropped from a search airplane and attached to the spacecraft by pararescue personnel, has been provided. This "collar" substantially increases the flotation life and stability of the spacecraft and provides a working base for the rescue team.

RECOVERY OPERATION EXPERIENCE

This discussion has thus far been directed primarily toward what is expected during forthcoming orbital flights. During the buildup phase of Project Mercury, however, several ballistic flights have been accomplished and a brief résumé of recovery experiences will serve to illustrate the types of operation which have been involved.

Figure 10 presents a typical distribution of recovery forces. A ship, such as an aircraft carrier or a smaller ship equipped with a helicopter landing platform, which has helicopters aboard, is stationed along with search airplanes in the high probability landing area. Ships and search airplanes are also positioned for support in the event of an overshoot or undershoot and, in addition, the launch-site forces are activated. The deployment of forces in this manner in support of ballistic flights serves two purposes: first, to recover the astronaut and spacecraft and, second, to evaluate recovery procedures planned for forthcoming orbital flights.

In the three Mercury-Redstone flights to date wherein this type of recovery-force distribution was used, one flight terminated within a few miles of the retrieval ship in the high probability landing area. A search airplane received the UHF/DF signals from the spacecraft at the time of main parachute deployment, homed on the descending spacecraft, and actually gained visual contact prior to the landing. The airplane, in turn, vectored a retrieval helicopter from the ship to the scene to effect recovery.

On the second flight which carried the chimpanzee, Ham, the spacecraft landed approximately 100 miles downrange of the planned landing area due to a launch-vehicle malfunction. Search airplanes were directed to the general landing area by the tracking-computing network indications. The airplanes then homed on the spacecraft UHF/DF signals and vectored helicopters from a ship based approximately 50 miles away to effect recovery. In this instance a destroyer was also on the scene.

On the recently completed manned flight, visual contact with the descending spacecraft was made by a retrieval helicopter airborne in the planned landing area. The helicopter followed the spacecraft down to the water and then effected retrieval.

In three Mercury-Atlas flights to date, one resulted in a landing approximately 500 miles short of the high probability landing area. Airplanes were initially directed to the general search area from information based on reentry sighting reports from the deployed ships. This area was subsequently verified by SOFAR information. The airplanes flew into the search area, established UHF/DF contact with the spacecraft, and vectored a ship from the secondary recovery area to effect retrieval approximately 8 hours after spacecraft landing.

Another flight landed close to the planned landing point, was located by a search airplane, which in turn vectored a helicopter in for retrieval.

A third flight was aborted in the launch-site area shortly after launch. The spacecraft escaped from the launch vehicle, descended on the parachute, and landed in the surf. Retrieval was effected in this instance by a helicopter; however, a LARC was also in the area for support.

In addition to the aforementioned operations, one Little Joe flight was retrieved by a destroyer in waves having an average height of about 10 feet and winds of about 35 knots indicating the ability to effect retrieval in some adverse-weather conditions. It is important that the spacecraft have a landing capability under these same conditions.

In over-all support of these flights, the recovery forces have had to exercise nearly all of the recovery equipment and techniques planned for spacecraft location and retrieval operations. In each instance, these systems have functioned adequately and recovery was accomplished within the time requirements previously specified.

TABLE I.- SPACECRAFT LOCATION AIDS

Type	Aid	Frequency, mcs	DF receiving equipment	Minimum expected life, hr	Operational range, nautical miles (altitude, 10,000 ft)	Remarks
Electronic	UHF SARAH Beacon	243	SARAH receiver (Simmonds, ITT)	24	40	Primary
	UHF Super SARAH Beacon	243	SARAH receiver (Simmonds, ITT)	24	60	Primary
	Survival-kit UHF SARAH Beacon	243	SARAH receiver (Simmonds, ITT)	24	50	Operated by astronaut after egress
	High-power UHF transceiver	Approximately 300	ARA-25, ECM, ITT any UHF receiver	12	60	Only one used at a time. DF capability lost if listening on UHF or using HF
	Low-power UHF transceiver	Approximately 300	ARA-25, ECM, ITT any UHF receiver	12	50	Only one used at a time. DF capability lost if listening on UHF or using HF
	HF SEASAVE Beacon	8.364	Land-based DF network (FCC and Navy)	24	1,200 (all times) 2,200 (selected times)	
Acoustic	SOFAR bomb		MILS network		Atlantic Ocean, California to Hawaii in Pacific Ocean	Depth setting - 3,500 ft (fixed SOFAR set for 4,000 ft)
Radar	Radar chaff	S, C, and X bands	Land, ship, and aircraft radar	$\frac{1}{2}$ to $2\frac{1}{2}$	100	Deployed at 21,000 ft, 300 sq ft Echo area
Visual	Fluorescein sea marker		Visual search	4	3 to 10 (dependent on light conditions)	
	Flashing light		Visual search	24	3 to 5	Useful only at night

RECOVERY AREAS

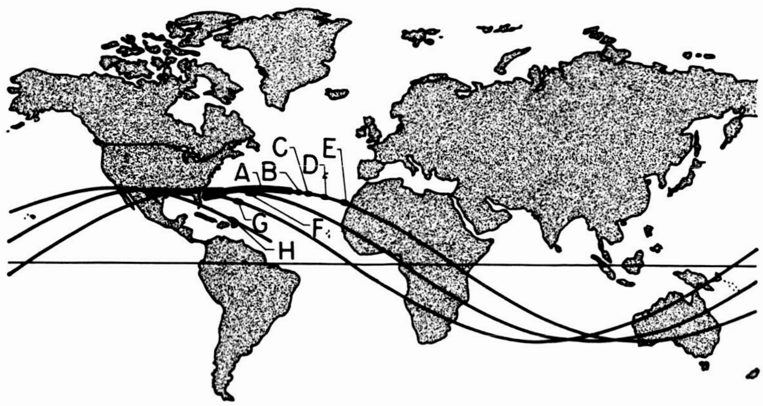


Figure 1

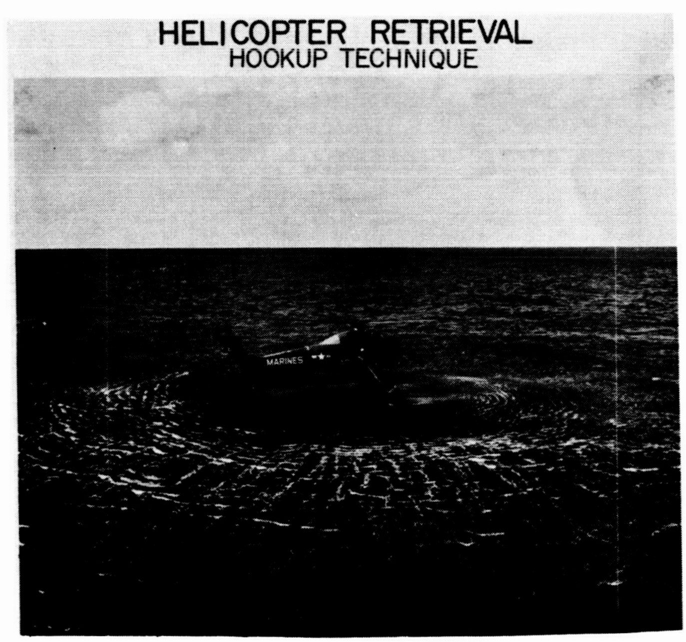


Figure 2 S-61-34

[redacted]

HELICOPTER RETRIEVAL
ASTRONAUT PICKUP

Figure 3 S-61-35

RETRIEVAL BY SHIP
TYPICAL HOOKUP

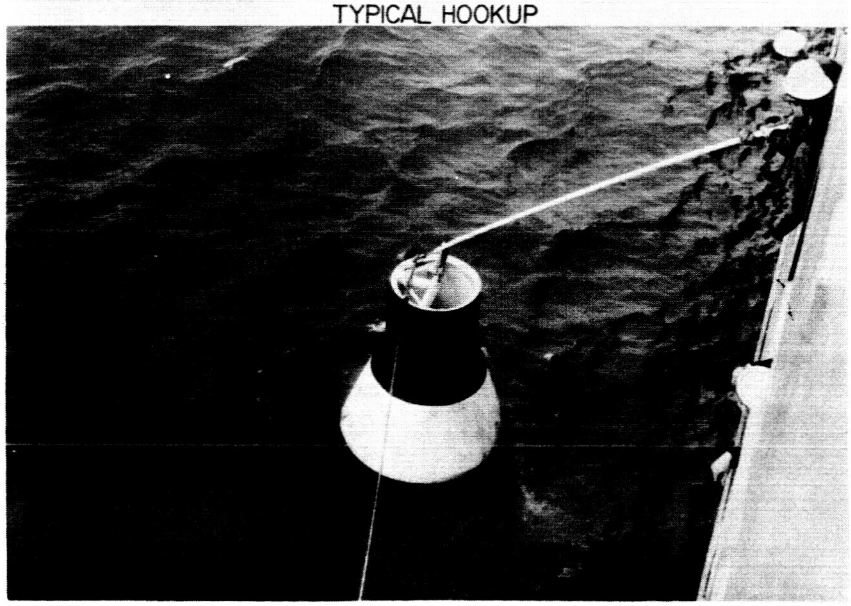
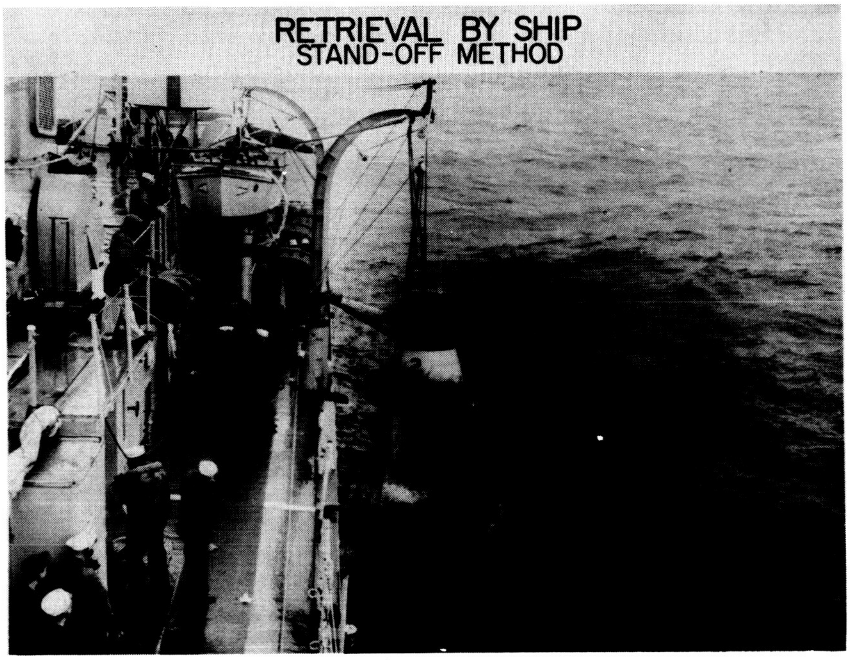


Figure 4 B-60-440

[redacted]



B-60-443

Figure 5

S-61-44

Figure 6

Figure 7 S-61-45

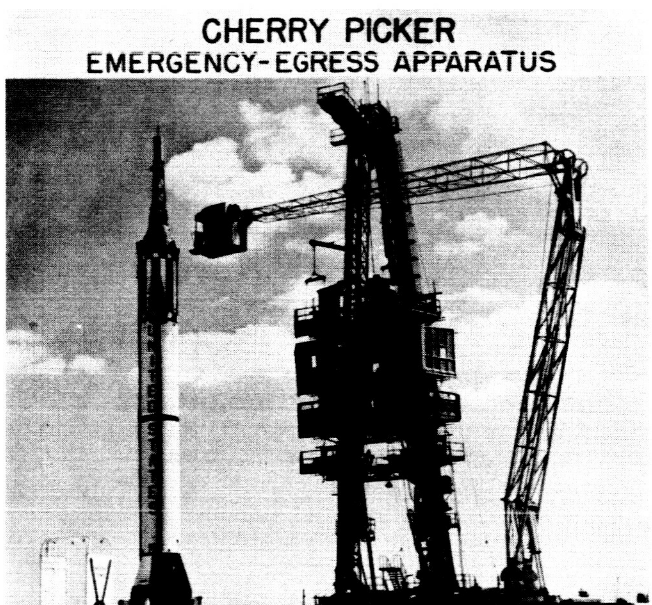


Figure 8 S-61-46

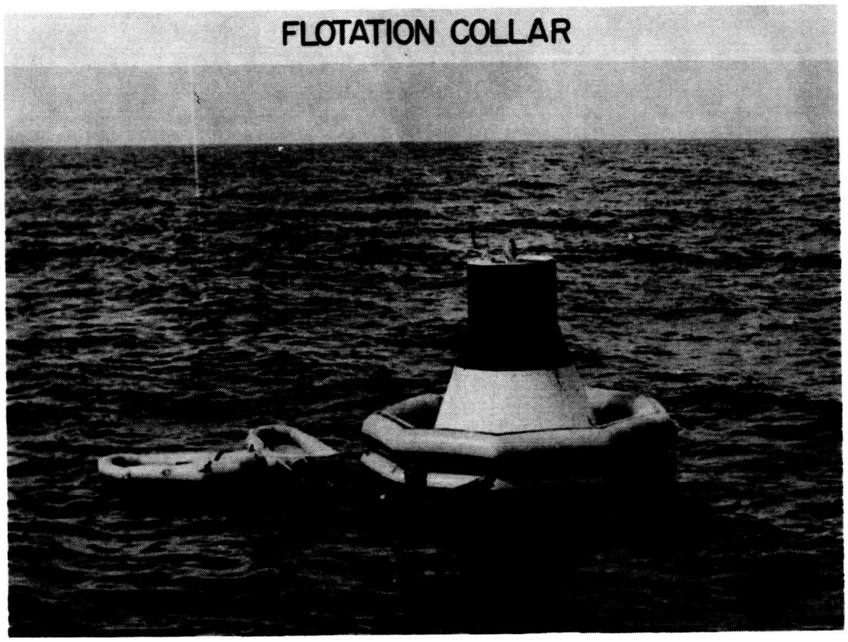


Figure 9 S-61-36

BALLISTIC FLIGHTS

TYPICAL RECOVERY AREAS FOR PLANNING PURPOSES

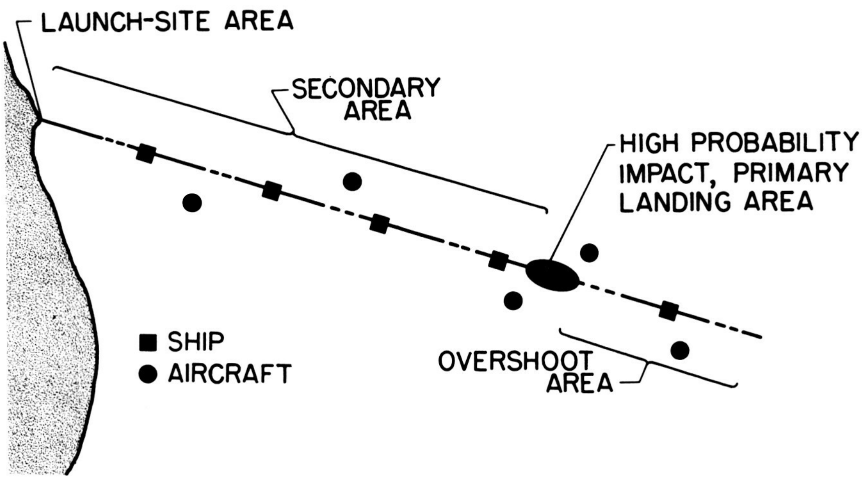


Figure 10

[redacted]

MERCURY FLIGHT SAFETY AND RELIABILITY

By F. J. Bailey, Jr., and John C. French

NASA Space Task Group

Probably the most important factor governing the schedule and cost of future manned space flight projects is the degree of flight safety and reliability required. Flight safety and reliability, although they are closely related, are not exactly the same thing. The important distinction between the two is illustrated in the following table:

Reliability		Flight safety
Spacecraft and launch vehicle	Escape system	Crew survival
0.50	0.998	0.999
.90	.99	
.999	0	

The first column shows a range of numbers for the overall reliability of a spacecraft—launch-vehicle combination. The numbers represent the probability that a given mission will proceed to completion without mishap. The second column shows a corresponding range of values of escape system reliability that produces the same overall flight safety, as represented in the third column by the probability of crew survival.

The point is that flight safety can be achieved either by building a high reliability vehicle with little or no provisions for escape, as in the case of a commercial transport, or by attaching a highly reliable escape system to an unreliable vehicle.

In Project Mercury, launch vehicles already developed for purposes other than manned flight were used, and the time schedule did not permit any substantial changes to these vehicles. Attempts to achieve a desirable level of flight safety through intense concentration on the escape system have been made but the probability of successful completion of a mission is lower than would have been chosen if the launch vehicles had been developed from scratch. For future manned systems the cost of a mission failure in terms of money, time, and national prestige, even though the crew survives, is so tremendous that an all-out effort must be made to achieve flight safety through overall reliability rather than through escape systems.

This reliability must be designed and built into the vehicle before it ever flies rather than achieved through successive modifications following a series of flight failures. The attitude that any flight failures are tolerable or are to be expected must not exist.

The rest of this paper deals with the reliability effort required and is confined to areas that stand out as being particularly important as a result of the experience on Project Mercury. These areas are indicated in figure 1.

The reliability effort involves three major areas: design, fabrication, and operation. In the first of these areas, an attempt is made to insure that a high degree of reliability is inherent in the basic design that is adopted. In the other two areas constant effort must be made to prevent degradation of that inherent reliability by human error in the fabrication and handling of the hardware.

In the design stage, there is needed a design using components that fail very infrequently combined in a system in such a way that the system keeps going in spite of any reasonable number of component failures. The Mercury spacecraft represents a determined effort to produce such a design. The requirement for selection of high reliability components for the spacecraft systems generated some very extensive development, qualification, and acceptance test programs. These test programs are discussed in a previous paper by André J. Meyer, Jr., William M. Bland, Jr., and Alan B. Kehlet. The problem of keeping systems functioning in spite of the inevitable component failures was handled by incorporating at least one order of redundancy into all critical systems. Wherever it could be done without serious penalty, an even higher degree of redundancy was introduced.

Some critical one-shot systems - that is, systems that act only once per flight - where redundancy has been incorporated in the design are listed in the following table:

(1) Tower and spacecraft separation
  Release
  Separating impulse

(2) Retrograde impulse

(3) Parachutes

The problem of being sure that the escape tower can be released from the spacecraft and that the spacecraft can be released from the launch vehicle has been handled by using clamping rings divided into three segments and held together by three double-ended explosive bolts. Firing any end of any bolt effects the release. The automatic system fires one end of each bolt from one electric circuit and the opposite end of two bolts from a completely independent circuit; an astronaut manually operated backup fires the opposite end of the third bolt through a percussion device and in addition sends electrical signals through the two automatic electric circuits.

Separating force is provided in the case of the tower by the escape rocket fired by a dual igniter with two independent initiation systems to each part. These systems include multiple squibs with independent circuits from different batteries. In the case of the spacecraft, separating force is provided by three solid fuel posigrade rockets with similar redundant firing circuits. Under normal conditions any one of the three posigrade rockets provides an adequate separation velocity.

For retroimpulse there are three solid fuel rockets with dual igniters fired by dual circuits. They may be initiated automatically or by either astronaut or ground command. Only two of the three retrorockets are required to effect a satisfactory reentry.

The primary parachute system is automatic. It incorporates dual barostats, dual power sources, and manual backup of each main function in the sequence. The entire automatic system is backed up by a manually operated reserve parachute system.

A number of critical systems of the spacecraft must operate continuously throughout the flight. The frequency of failure of components in these systems is in general proportional to the length of time they are operated and hence to the length of the mission. Some of these systems are power supply, environmental control, communication, and attitude orientation.

The direct-current power supply incorporates redundancy in the form of three main batteries, two standby batteries, and an isolated battery for critical functions. The alternating-current power system incorporates two main inverters backed up by a standby inverter capable of replacing either one.

The environmental system incorporates the basic redundancy of a full pressure suit in a controlled cabin environment. Manual controls are provided to back up the automatic-control functions. An emergency oxygen supply is available to the suit as a further backup in the event of simultaneous malfunctions in both suit and cabin controls.

The communications-system redundancy includes dual command receivers, high- and low-frequency telemetry transmitters, dual UHF voice transmitters and receivers backed up by an HF transmitter-receiver, and a telegraph key operating through one of the telemetry transmitters.

The attitude orientation system which is particularly critical for retrofire consists of a primary automatic system backed up by an independent manual system.

The automatic and manual backup system for spacecraft orientation is shown in figure 2. The automatic branch of the system includes the horizon scanner to sense spacecraft attitude, the autopilot to generate the necessary attitude and rate signals, and solenoid valves to control the flow of hydrogen peroxide from the automatic peroxide supply tank to the automatic system thrust chambers. These thrust chambers provide the actual control moments in roll, pitch, and yaw.

In the manual branch of the system the pilot determines attitude from either instrument or visual indications. By means of a manual control stick he can control attitude through three alternate channels. He can send electrical signals to the solenoid valves of the automatic system. He can mechanically operate the proportional valves of the manual system. These valves regulate the flow of hydrogen peroxide from the manual supply tank to the manual system thrust chambers. In the third channel he can send electrical rate command signals through a rate damping system to solenoid valves in the manual peroxide system. The variety of control modes available is greater than absolutely necessary for reliability. It was incorporated partly for future use in research on space attitude control requirements.

One general problem of significance in connection with future spacecraft designs has become apparent from study of these time dependent systems. Where reliability is provided, as in Project Mercury, by the redundancy of a primary and a backup system, the mission rules stipulating conditions for discontinuing the mission become critical in establishing mission completion reliability. If the mission rules require abort following failure of the primary system, then as far as mission completions are concerned, only the reliability of the single primary system is realized and not the much greater reliability of the redundant system. For much longer missions in the future, it will be very difficult to design any single system with sufficient reliability. The only practical solution may turn out to be in-flight repair or triple systems with the redundancy of the first two used for a high probability of mission completion and the third used for the infrequent abort.

In the old problem of keeping the inherent reliability from being degraded in hardware fabrication and operation, the things that stand out in the experience with Project Mercury are indicated along the bottom row of figure 1.

Quality control monitoring all the way back to the parts suppliers is a necessity. Preflight functional checks are a completely unacceptable substitute for built-in quality and rigorous inspection. Every failure must be analyzed and followed within days, not months, by intelligent directive action.

Feedback of operational experience into fabrication is recognized as important everywhere. It must be introduced as early as possible, must carry the weight of top management behind it, and must produce prompt response so that the required changes are accomplished in the next vehicle and not the 17th down the production line. To get this operational feedback into Project Mercury, a succession of Development Engineering Inspections were conducted with the inspection team heavily populated with operations personnel and the Inspection Board chaired by the Operations Director.

With different groups responsible for the launch vehicle and the spacecraft, there is need for very special planning and procedures to insure proper handling of interface problems. It has been found necessary in the field to establish the joint inspection team charged with the responsibility for witnessing all mating and other interface activities for measuring and verifying the adequacy of all physical clearances, inspecting all structural joints and electrical connections, and assuring that no undesirable debris is left in critical areas when they are finally buttoned up. Adequate access ports for field inspection are also an absolute requirement.

Special procedures have been set up for maintaining and periodically distributing one and only one official interface wiring diagram, reflecting the exact current status of the wiring on the vehicle at specified dates.

While these field procedures appear to take care of safety aspects adequately, there probably is a need in any future project for additional measures to prevent interface clearance problems from ever reaching the field.

Last but by no means least, is the problem of determining that the vehicle is in fact ready for launch. In Project Mercury, the philosophy has been adopted that the spacecraft will not be launched with any observed difficulty unexplained or uncorrected. To insure that this philosophy is carried out, a series of preflight review meetings are held in which all malfunctions observed in the system and all changes and corrections made are discussed in detail with the specialists responsible for the check-out of each system. These meetings are quite lengthy and go quite deeply into technical detail with very free and frank discussions of each problem.

These detailed meetings on the major pieces of equipment are, of course, followed by a final mission review meeting in which all elements involved in the mission provide a final confirmation of their readiness. In the prelaunch reviews much effort has been expended to make certain that at the time of launch there is no question in the mind of anyone involved in the mission as to the exact status and prior history of all systems involved.

In summary, the items that stand out as important for future manned space-flight programs are:

(1) A basic design predicated on recognition of the fact that a very high degree of inherent reliability in both spacecraft and launch vehicle is an overriding requirement

(2) A special effort to realize the full inherent reliability in the field involving

(a) An intensive quality control effort aimed at minimizing the need for reliance on last-minute functional checks

(b) Early and vigorous feedback of operational experience

(c) Special attention to interface areas

(d) Adequate machinery to insure full attention to all prelaunch difficulties, plus a rigid policy of no-launch with any prelaunch difficulty unexplained or uncorrected

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FLIGHT SAFETY AND RELIABILITY

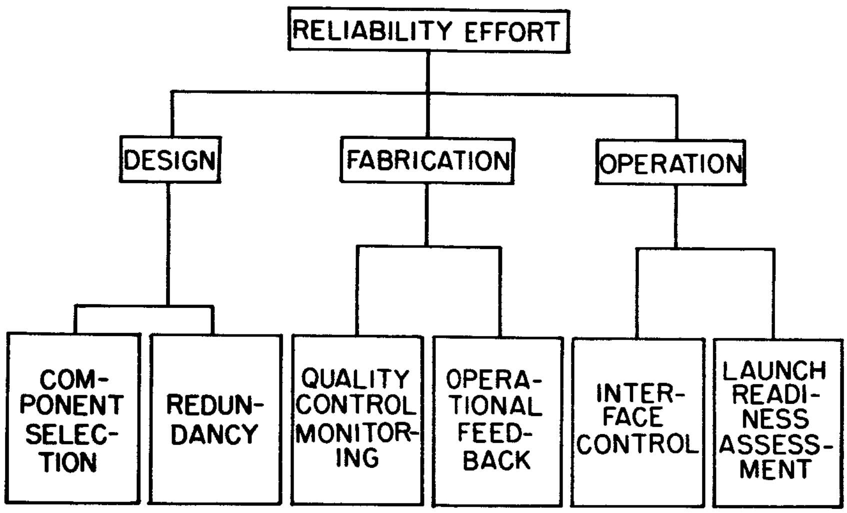

Figure 1

FLIGHT SAFETY AND RELIABILITY
ATTITUDE ORIENTATION

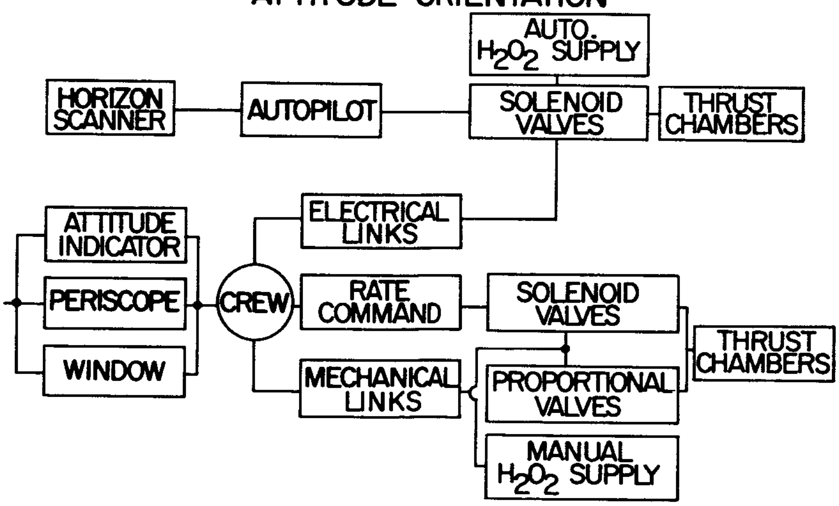

Figure 2

[redacted]

MERCURY CREW TRAINING DEVICES

By Harold I. Johnson

NASA Space Task Group

INTRODUCTION

The purpose of this paper is to present the important astronaut training devices being used in Project Mercury, to point out how these devices were used to help solve the crew participation problems, and to forecast what similar devices will be required for Project Apollo.

The two basic objectives of Project Mercury are (1) to put man into space and return him safely, and (2) to determine how man performs in space. The crew participation requirements are a direct outgrowth of these two objectives. For example, in order to expect the astronaut to return safely he must be given thorough training in normal spacecraft systems operations so that he will recognize immediately any potentially dangerous systems failures that may occur. He must also know how to restore these systems to acceptable operation by use of the manual backup controls, if necessary. Finally, he must be trained thoroughly in manual control of spacecraft attitude, particularly during firing of retrograde rockets; a gross failure in this particular task (if it has become necessary for the astronaut to take manual control) would almost certainly be disastrous. Failure to maintain control of spacecraft attitude would be disastrous because the spacecraft would then probably stay in orbit for an excessively long period of time, so that there would be great danger of completely depleting the onboard oxygen. Even if the oxygen were not completely depleted, the very shallow reentry angle expected during the eventual free reentry would probably cause the interior temperature of the spacecraft to exceed the limits for human survival. The second basic objective, that of determining how man performs in space, suggests that the astronaut should receive training as a spacecraft engineering test pilot and also training as a navigator-observer.

The primary objective in the selection of the particular astronaut training devices was to provide training in one or more of the areas just mentioned; the secondary, though still important, objective in using many of these devices was to familiarize the astronauts with the physiological stresses to be expected in the flights and, as a consequence, to allow the astronauts to establish confidence in their own ability to withstand any and all stresses that might be encountered.

TRAINING DEVICES

The following list indicates the astronaut training devices to be discussed:

1. Controls Simulator No. 1	
2. Controls Simulator No. 2	Fixed-base trainers
3. Mercury Procedures Trainers	
4. Pilot-Egress Trainer	
5. MASTIF Trainer	
6. Johnsville Human Centrifuge	Moving-base trainers
7. Orbital Attitude Trainer	
8. Environmental Controls Trainer	
9. Flight Instrument Display Mock-Up	Indoctrination devices
10. Ground-Recognition Trainer	

This is not a completely comprehensive list of all the devices that have been used; for example, it leaves out such devices as the disorientation device at Pensacola, Fla. and the airplanes outfitted for zero-g familiarization. The Mercury Procedures Trainers, the Johnsville Human Centrifuge, and the Orbital Attitude Trainer are considered to be by far the most important three training devices. (Incidentally, Commander Shepard concurs in this evaluation.) The following discussion will indicate how these devices are used to solve crew participation problems in Project Mercury and will indicate the probable application of similar devices to Project Apollo.

Fixed-Base Trainers

Figure 1 shows the Controls Simulator No. 1, which made use of an Electronic Associates Analog Computer. This elementary trainer had an airplane pilot's seat, a rudimentary instrument panel containing approximations of the Mercury rate-and-attitude indicators, an indicating accelerometer, an altimeter, and a research-type three-axis hand controller. The primary purposes of this training device were to test the feasibility of manually controlling the spacecraft attitude in simulated space flight, to test the feasibility of holding the spacecraft in a prescribed attitude during simulated firing of the retrorockets, and to test the feasibility of stabilizing and controlling the anticipated oscillations of the spacecraft during simulated reentries. Tests with this simulator indicated that the pilot would be able to control the Mercury spacecraft satisfactorily in all these necessary flight phases by using either of the two different manual reaction control systems provided. This simulator also introduced the manual control tasks to the seven Mercury astronauts in May of 1959. Simple fixed-base simulators of this type are considered to be extremely valuable in early phases of manned space-flight projects. Many setups of this type will undoubtedly be used in Project Apollo.

Figure 2 shows a view of the Controls Simulator No. 2, which made use of a modified Melpar MB-3 computer. This trainer was very similar to the first one discussed except that it featured an early space couch instead of a pilot's seat and it had a prototype three-axis Mercury hand controller instead of a research-type hand controller. Note that a set of rudder pedals was provided for controlling yaw motions when the yaw freedom of the hand controller was locked out. Several crew participation problems were studied with this fixed-base trainer, such as (1) the feasibility of manual control with pressurized pressure suit, (2) the advantages and disadvantages of airplane-type rudder pedals for yaw control, and (3) the effect of retrorocket firing sequence on astronaut ability to hold retro-attitude precisely.

Figure 3 shows one of the two Mercury Procedures Trainers; the one located at Langley Field, Va. (The other trainer is located in the Mercury Control Center at Cape Canaveral.) Note the capsule in the background and the instructor's console in the foreground. This particular trainer provides active simulation of all of the approximately 20 combinations of manual and automatic spacecraft attitude and/or rate control modes available in the Mercury spacecraft insofar as the rate-and-attitude indicator is concerned. The trainer itself does not move. A simplified periscope display driven by the computer is also provided. More important than these, however, is the fact that all the primary spacecraft systems are simulated electronically or mechanically, and approximately 276 separate failures can be introduced by the instructor into the trainer at various times during simulated missions. Repeated exercising of this capability allows the astronauts to become very proficient in actuating the many manual backup controls provided in the spacecraft, and therefore greatly improves chances for mission success and astronaut safety. In Project Apollo, the same extensive participation of the flight crew should be exploited, perhaps to a greater degree than in Project Mercury. The reason that the extensive flight crew participation should be exploited in Project Apollo is simply that the length of the missions will be much greater so that the inherent reliability of the automatic systems will tend to be less. On the other hand, because there will be three human backups in Apollo instead of only one, some further improvement in the chances for mission success and astronaut safety might be expected because of human participation. Perhaps because these procedures trainers have such tremendous direct implications relative to astronaut safety, they are generally considered to be the most valuable of all the Mercury trainers used. Accordingly, heavy emphasis on this type of trainer can be expected in Project Apollo.

There is a reason for having two procedures trainers in Project Mercury. This point has been mentioned briefly in a previous paper by Christopher C. Kraft, Jr., and C. Frederick Matthews. The trainer at Langley is connected to a simulated remote-site flight controllers' console which allows joint training of the remote-site controllers and the astronauts. This trainer also is used to prepare the astronauts for the acceleration training programs at Johnsville, Pa., which will be discussed subsequently. The trainer at Cape Canaveral is connected to the actual Mercury Control Center consoles and thereby allows joint training of the Control Center flight controllers and the astronaut. The Cape Canaveral trainer is also used for intensive preflight training of the astronaut who will man the next mission. One aspect of this training is concentration on the specific manual capsule maneuvers that are to be performed in the next particular flight. Using the two procedures trainers in these ways has been found to be very effective, and similar applications of the Apollo procedures trainers should be considered.

Moving-Base Trainers

Figure 4 shows the Pilot-Egress Trainer. This trainer was a mock-up of the Mercury spacecraft, in which the hydrodynamic stability and the escape-path obstructions of the actual spacecraft were essentially duplicated. This trainer was used to develop the preferred method of astronaut recovery following a normal mission and to train the astronauts in emergency escape from the spacecraft. In the latter category, the astronauts practiced escape from the spacecraft through the top hatch (as shown in fig. 4) and escape through the side hatch when the spacecraft was either floating or submerged. Though simple, this device was found to be very valuable, not only for the engineering development and astronaut training aspects it provided, but also because it was a good confidence builder for the astronauts. It is too early to predict whether any formal egress training will be required in Apollo; if so, it is hoped that actual Apollo flight vehicles can be used for training.

Figure 5 shows the Multiaxis Spin Test Inertia Facility (MASTIF) Trainer of the NASA Lewis Research Center. This MASTIF Trainer is a disorientation familiarization trainer. The astronauts were whirled up to rotational speeds beyond those at which disorientation occurs. They then practiced stopping the violent rotational motions by use of the reaction control system and the Mercury angular rate indicator provided. This simulator was valuable because it indicated that if a Mercury spacecraft ever inadvertently tumbles, it is extremely likely that the pilot will be able to save himself providing the rate indicator and at least one of the two manual control systems are still functioning. In Apollo, it is doubtful that this type of training will be necessary, inasmuch as all astronauts successfully stopped the violent motion the first time they tried. On the other hand, if such a facility exists during Apollo development, it would seem advisable to make use of this facility because of confidence-building effects on the flight crew.

Figure 6 shows the Johnsville Human Centrifuge. Three different month-long acceleration training programs have been run so far at this facility. The first program, in August of 1959, was an engineering feasibility validation of the overall Mercury concept. Very high peak g Atlas aborts (up to 18g) were tried. It was shown that even if the astronaut was unable to actuate controls at these extreme g levels (which fortunately are of only very short duration) at least he did not usually black out completely. In general he was able to exert effective manual control of spacecraft attitude and to actuate backup controls for systems failures for most of the periods covered by the overall g pulses. The second program, in April of 1960, investigated the problems of crew participation during both Atlas and Redstone acceleration profiles when the normally reduced cabin pressure and an abnormal, pressurized pressure suit were combined with the acceleration stresses. The third program, in October of 1960, was an intensive indoctrination for Redstone missions. This program was covered in detail in a previous paper by Richard S. Johnston and Gerard J. Pesman. A similar program aimed at Atlas indoctrination is scheduled for next month. All these programs contributed heavily to a better understanding of how personnel equipment and cockpit layout affect possible crew participation during the critical periods of high acceleration which occur at times during space flights. Needless to say, the acceleration training programs are considered invaluable in the Mercury project and should prove to be equally so in the Apollo project.

Figure 7 shows the orbital attitude trainer. This figure shows a closeup view of an early configuration of this trainer in use; note the simulated periscope earth-path display. Figure 8 shows a bottom view of the current configuration. The trainer consists basically of a "flattened out" Mercury-type couch supported on a 5-inch-diameter spherical air bearing. For orientation purposes, one should compare this trainer to a Mercury spacecraft on end with heat shield down. The trainer is free to move $\pm 35^{\circ}$ in pitch and yaw and can be rolled an unlimited amount. When in use, the trainer is supported by a hemispherical sheet of compressed air approximately 0.001 inch thick within the spherical air bearing. This nearly frictionless support results in almost no resistance to rotation about any axis except resistance due to inertia, which, of course, is the only effective resistance offered to spacecraft rotation when in space. Also, when in use, the trainer-plus-astronaut combined center of gravity is made to fall at the center of the spherical ball of the air bearing, so that the trainer has the characteristic of zero centering forces or zero static stability about any axis - also representative of space flight. The trainer has either a simulation of, or the actual visual reference system of, all three spacecraft visual reference systems, that is, the periscope, the spacecraft rate-and-attitude indicator, and the spacecraft window. A moving earth scene utilizing a 10-foot-diameter back-projection screen is provided on one wall for viewing through the simulated periscope (this display is shown in fig. 7), and a lighted horizon is provided at the proper place in the ceiling of the room for viewing out the window. Both the manual proportional and the manual fly-by-wire attitude control systems of the spacecraft are simulated. Retrorocket disturbance jets are also provided. All jet systems are powered by compressed air taken onboard by means of a hole through the center of the air bearing. The types of training provided with the trainer include practice in precise attitude stabilization for long periods of time in orbit, practice in making precise single- and multiple-axis maneuvers from one specified attitude to another, and practice in holding spacecraft retrograde attitude precisely in the face of programed torques representing torques expected as a result of retrograde rocket firings. These tasks are done first by using each visual reference system alone, and then by using all the reference systems in various combinations. The trainer is considered quite valuable in the Mercury program because it is the only trainer which includes astronaut bodily rotations together with all the available visual display systems which can be used simultaneously. This trainer does, however, present the subject with incorrect body-pressure cues arising from the necessity of working in a 1g field. In spite of this inaccuracy, it is believed such a trainer will probably find application in preparing the Apollo flight crews.

Indoctrination Devices

Figure 9 shows the Environmental Controls Trainer. This device has been discussed in a previous paper. It was found more valuable for equipment development than for astronaut training. It is unlikely that a device of this type will be used for training in Apollo. Experience indicates the kind of training provided by such a device can be obtained better in procedures trainers and in actual spacecraft mounted in low-pressure chambers.

Figure 10 shows the Flight-Instrument-Display Mock-Up. This is a one-half-scale rough transparent model of a Mercury spacecraft mounted within a four-gimbal stand so that the mock-up can be readily turned to any attitude. The mock-up contains a Mercury rate-and-attitude indicating system without the horizon scanner. The cases of the attitude gyroscopes have been removed so that the trainee can observe how and why the attitude gyroscopes tumble when the spacecraft is rolled too far. This device is used to teach the astronauts how the attitude indicating system can be lost as a result of large-amplitude maneuvers, how the indicating system can be regained if it has been lost, and how the attitude indicators can read falsely as a result of gyroscope cross coupling at times when the floating gyroscope axes are not orthogonal with the spacecraft axes. Although this device is very useful in Project Mercury, it is hoped that the need for such a device will not exist in Apollo because of the application of more advanced attitude gyroscopes in the Apollo vehicles.

Figure 11 shows the Ground-Recognition Trainer, which consists of a couch, an actual periscope, a back-projection screen, and a motorized slide projector. The slide projector puts a colored, moving image of the earth on the screen. This image is viewed through the periscope which is situated at a proper distance from the screen to simulate the geometry of a real spacecraft periscope aimed at the real earth from orbital altitude. The endless film strip, provided by the Aeronautical Chart and Information Center of St. Louis, Mo., contains the earth scenes to be expected during a standard three-orbit Mercury mission. The obvious purpose of this trainer is to familiarize the astronaut with the periscope scenes to be expected and to train him for his role as navigator-observer.

CONCLUDING REMARKS

The primary training devices used in Project Mercury have been discussed, and ways have been indicated in which many of the problems of crew participation were solved by using these training devices and simulators. Limited flight experience indicates the astronaut can operate at least as effectively at zero g in the actual flights as he can under 1g in the training devices. With reference to performance under increased g during launch and reentry, the astronaut can exert active control at least as well in the actual capsule as he can in a centrifuge, and in addition, he prefers the more purely linear acceleration characteristic of the actual vehicle as compared with the acceleration characteristic of the centrifuge. At this point, therefore, there is every reason to expect that the Apollo crew will be able to participate extensively in direct manual control of Apollo vehicles and in the direct management of Apollo onboard systems.

With reference to Apollo simulator requirements, there will probably be a need for several single-mission-phase fixed-base feasibility simulators, one or more fixed-base procedures trainers (at least one of which should incorporate a high-fidelity heavens display), a centrifuge training program, an orbital-attitude-type trainer, and possibly several smaller indoctrination or familiarization devices such as those found useful in Project Mercury.

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Figure 1

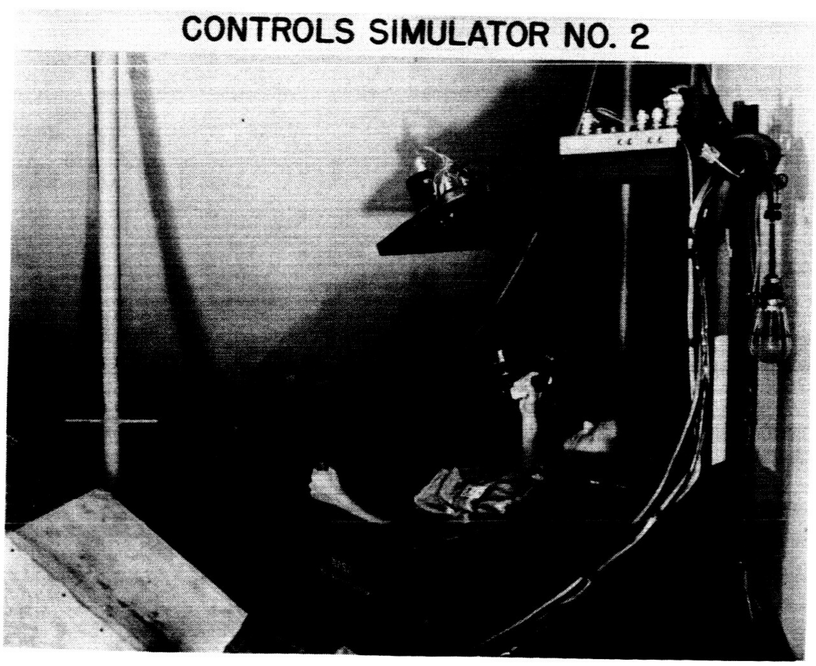

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Figure 2

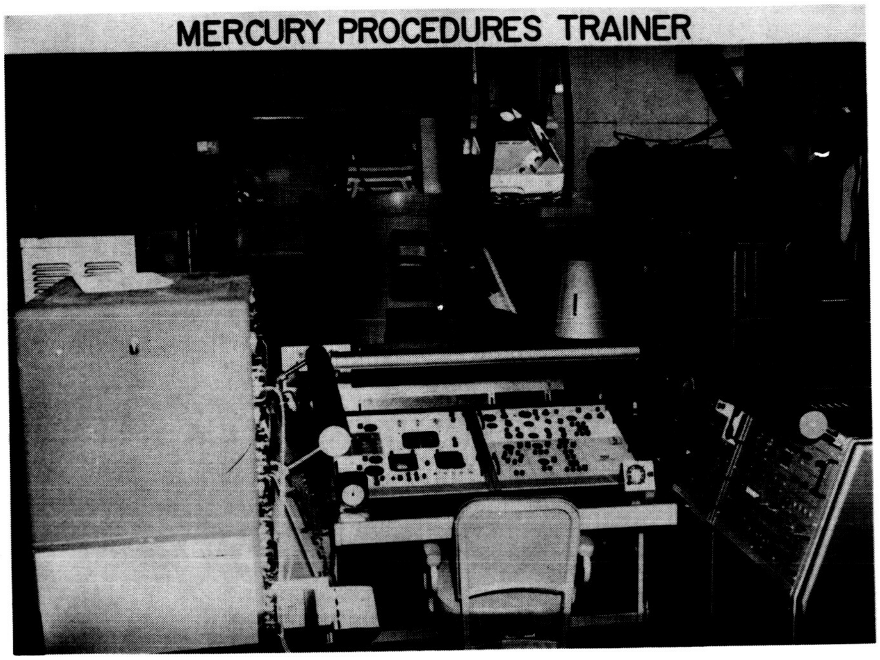

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Figure 3

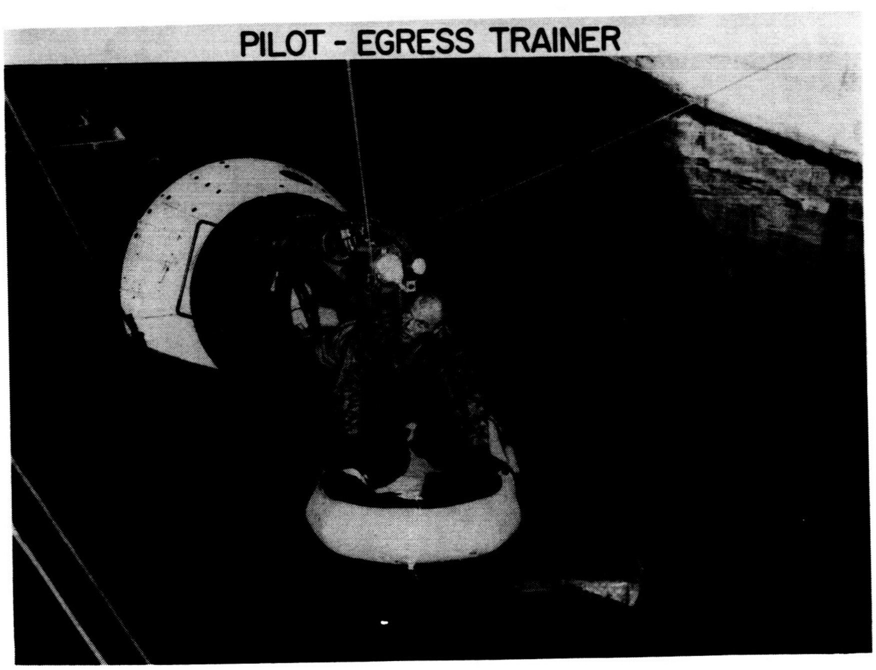

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Figure 4

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MASTIF SIMULATOR

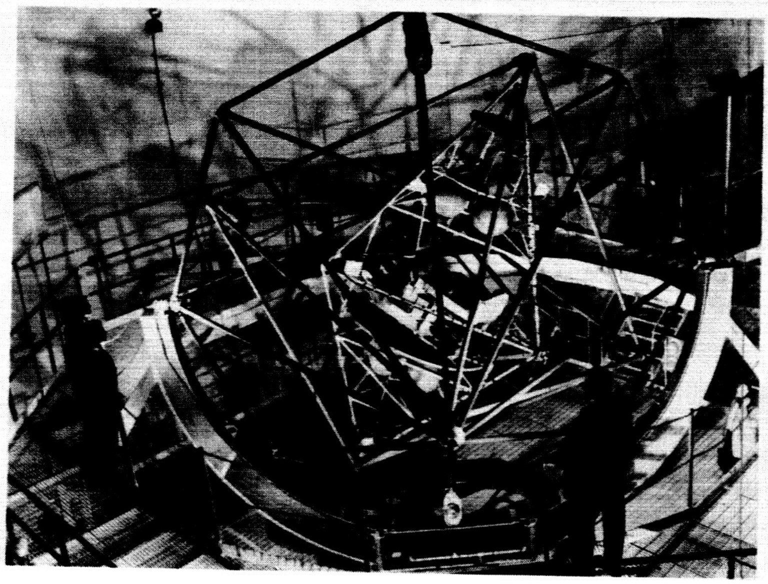

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Figure 5

JOHNSVILLE CENTRIFUGE

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Figure 6

EARLY ORBITAL ATTITUDE TRAINER

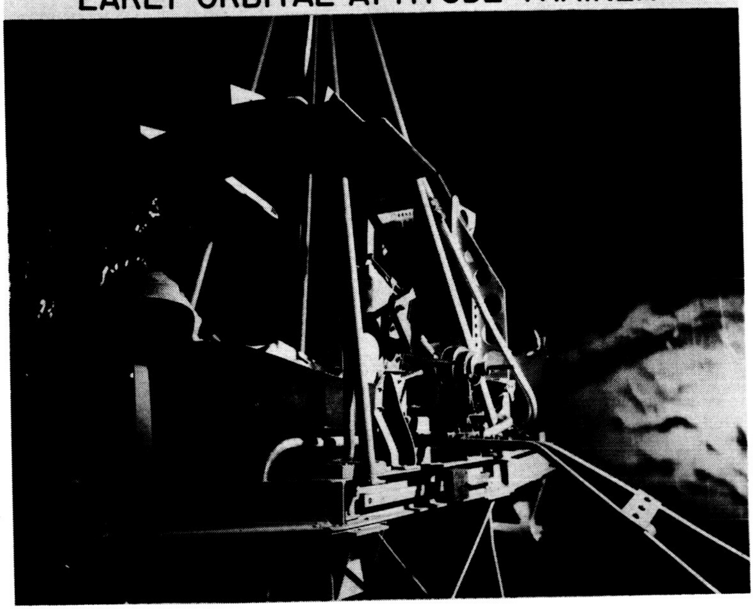

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Figure 7

FINAL ORBITAL ATTITUDE TRAINER

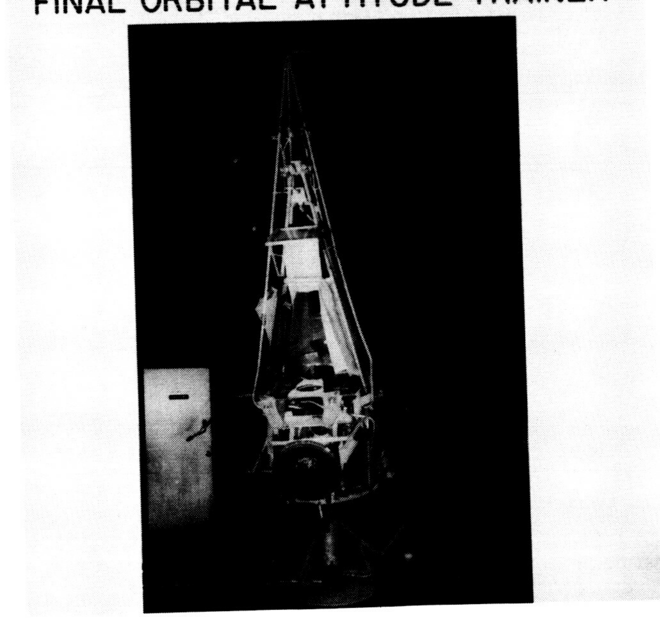

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Figure 8

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ENVIRONMENTAL-CONTROLS TRAINER

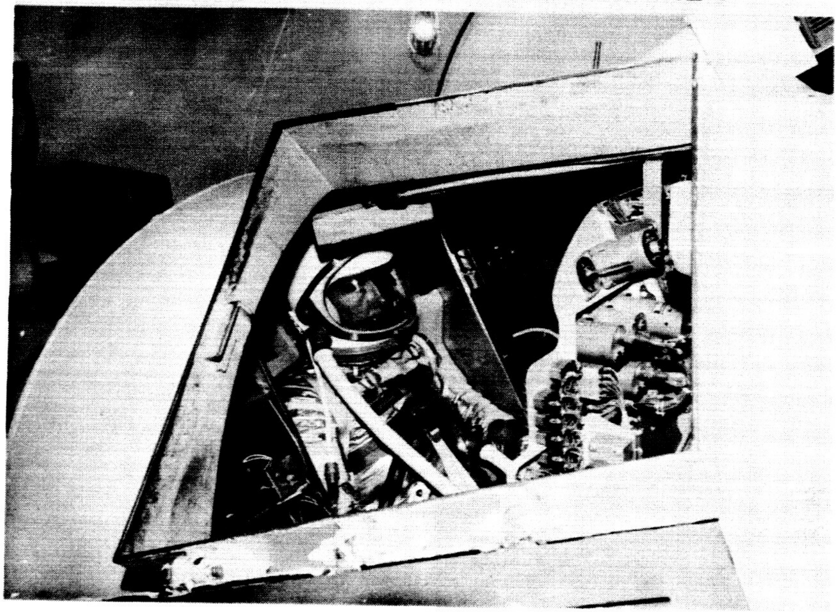

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Figure 9

FLIGHT-INSTRUMENT-DISPLAY MOCK-UP

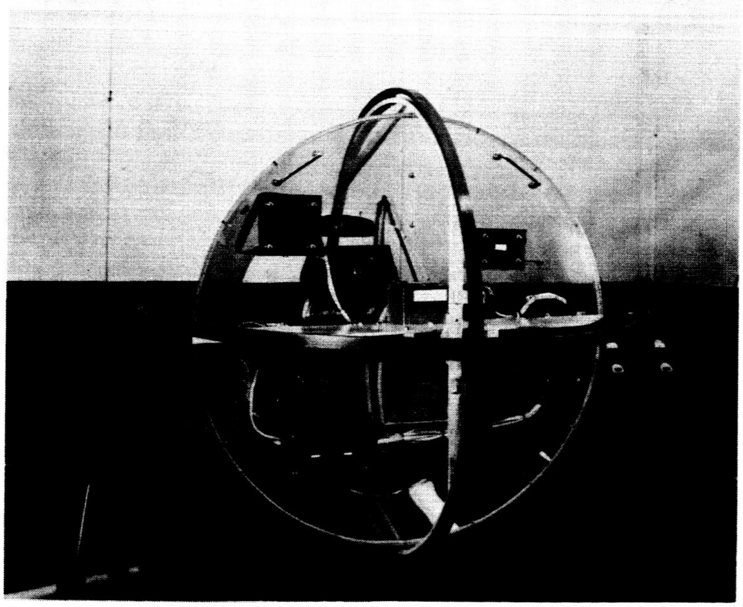

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Figure 10

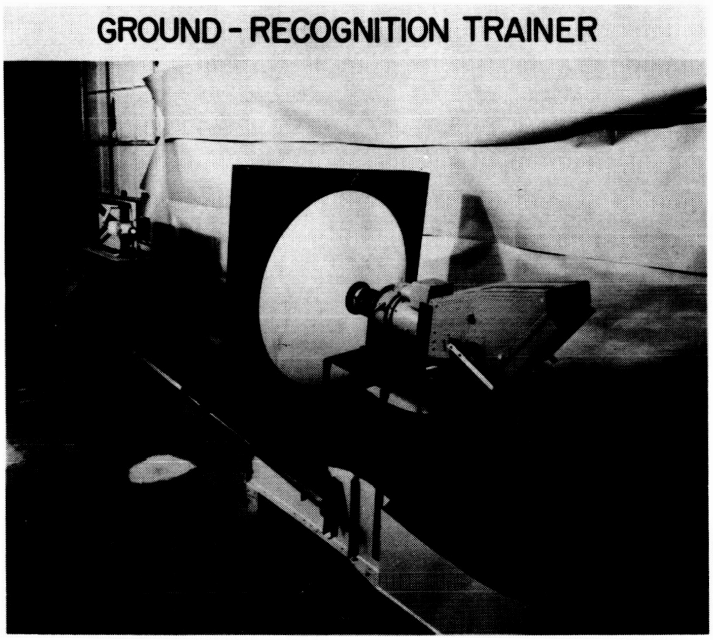

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Figure 11

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